INSTRUMENTATION OF A CESSNA 310H AIRCRAFT FOR ACADEMIC INVESTIGATION OF FLYING QUALITIES AND PERFORMANCE CHARACTERISTICS

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THESIS

INSTRUMENTATION OF A CESSNA 310H AIRCRAFT
FOR
ACADEMIC INVESTIGATION OF FLYING QUALITIES
AND
PERFORMANCE CHARACTERISTICS

by

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ABSTRACT

The two-course study of flight evaluation techniques offered by the Aeronautics Department of the Naval Postgraduate School requires an airborne laboratory phase of instruction that introduces the student to the actual problems encountered in obtaining accurate in-flight data. To satisfy this need, a civilian registered Cessna 310H aircraft, N164X, was leased by the Naval Postgraduate School in April, 1973. An airborne data acquisition system was designed and installed that allows three students to obtain individual measurements of twelve performance and stability and control parameters. The measurements are obtained using both electrical and differential pressure sensors, and are manually recorded by each student. Due to time constraints, no in-flight evaluation of the system has yet been conducted.

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The actual fabrication and installation of the many components of the instrumentation system were done by Aeronautics Department technical personnel under the direction of Mr. Robert Besel and Mr. Theodore Dunton. These included Mr. Cecil Gordan, Mr. Stan Johnston and Mr. Glenn Middleton. The authors are especially grateful for the assistance of Mr. Burtis Funk, who installed all of the instrumentation components.

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I. INTRODUCTION

The Aeronautics Department of the Naval Postgraduate School offers a two-course sequence in the study of flight evaluation techniques as a part of the graduate program in Flight Mechanics. The first provides an introduction to the methods and techniques involved in aircraft data acquisition and concentrates on performance testing. The follow-on course concerns itself with aircraft flying qualities and stability and control.

Prior to January, 1970, both of these courses used aircraft assigned to the Naval Auxiliary Landing Field, Monterey, California, to provide in-flight testing and data collection experience in the laboratory portion of the sequence. When Combat Readiness Training flying was discontinued at Monterey in January, 1972, actual airborne testing was removed from the flight evaluation course syllabus. The subsequent degradation of course content provided an extra impetus to find a means of operating an aircraft solely as an instructional laboratory vehicle of the Aeronautics Department. After careful study of the various problems involved in such operation, a civilian registered Cessna 310H aircraft was acquired by lease through funding provided by the Naval Air Systems Command.

Evaluation of flying qualities requires aircraft instrumentation in excess of that normally included in operational aircraft in order to record the rapidly changing parameters

associated with flight test maneuvering. A system was required that provided accurate, comprehensive measurements and readouts of the desired parameters with but minor modification of the aircraft. Such a system was designed that not only provided appropriate data acquisition, but also ensured maximum utilization of the aircraft as an instructional aid by designing into the system multiple readouts of each measured parameter so that on each flight three students could separately acquire simultaneous measurements of control forces and position, angles of attack and sideslip, normal acceleration, and altitude and airspeed.

Initial installation of this system was accomplished at the Naval Postgraduate School, Monterey, California, during May and June, 1973.

II. BACKGROUND

The Aeronautics Department's need for an aircraft to be used as a flying laboratory was recognized as far back as 1966. Professor C. D. Perkins of Princeton University, in the report of the 1966 Aeronautics Visiting Committee, stated: "One of the surprising aspects of the (Naval Postgraduate School Aeronautics) program is the lack of an adequate graduate aero sequence in the broad and important area to the Navy of flight mechanics, including advanced courses in stability and control, performance, design, etc." Since then, the program to acquire an aircraft for airborne laboratory and research projects has closely coincided with the development of the present two-course sequence in flight evaluation techniques as a part of the Flight Mechanics curricula.

In order for the flight evaluation sequence to be of any practical use, a means must be available of coordinating the theoretical principles developed in the classroom with the actual problems faced by the test pilot or engineer in acquiring data. Initially, North American T-28's were used to provide simple airborne data measurements. In 1968 permission was obtained from Commander, Naval Air Forces, Pacific Fleet, to instrument a Grumman US-2A aircraft. This instrumentation system was to provide sufficient sensor inputs to utilize the US-2A as either an airborne laboratory or as a testbed for research projects. The data acquisition system originally

installed in BUNO 136533 was refined and expanded and performed quite satisfactorily until the aircraft was damaged in an unintentional wheels-up landing in March, 1971.

During the period of April, 1971, to December, 1972, various Navy aircraft assigned to the Naval Auxiliary Landing Field, Monterey, were used to provide elementary airborne data acquisition, although no permanent flight test instrumentation was installed in any of the aircraft. With the closing of the Naval Auxiliary Landing Field and the discontinuance of CRT flying at Monterey in January, 1972, even this limited capability was removed. The flight evaluation sequence was subsequently taught during 1972 without an actual airborne laboratory phase; instead, precomputed problems were introduced as laboratory exercises.

The inherent deficiencies of this method, along with the recommendations of the Aeronautical Engineering Review Boards of 1969, 1970, and 1971, sparked the investigation of the means of obtaining a light twin-engine aircraft for use in the courses. Initial plans were to acquire a surplus United States Air Force U3A, the military version of the Cessna 310. However, funding limitation and the need for major overhaul of such aircraft made this approach impractical.

Funding support from the Naval Air Systems Command within AIRTASK A3203200/186B/3F41-421-200 dated 2 August 1972 was made specifically for the lease of a light twin-engine aircraft for use in the Flight Evaluation Techniques Program at the Naval Postgraduate School. Specifications for the aircraft

were drawn, bids were solicited, and after several delays, a contract was signed with McDonnell Enterprises of Mojave, California. Terms of the one-year, optional renewal contract cover the total cost of the operation of the aircraft for 250 flight hours per year. The aircraft, Cessna 310H side number N164X, was delivered to the Monterey Peninsula Airport 14 April 1973.

After familiarization flights totaling 11.5 hours, the aircraft was grounded until all aspects of its operation were reviewed and approved by the school administration. Although initial installation of the instrument system was begun during the grounded period, flight testing and evaluation of individual system components were impossible. Serious delays in the delivery of vital parts ordered through the Navy supply system also caused setbacks in the pace of the project. Subsequently, a full evaluation of the operation of the complete system was impossible before the date of this report.

III. REQUIREMENTS

In the engineering sense, the basic problem was to design, install, and evaluate a data acquisition system not only capable of providing sufficient information to evaluate the flying qualities of the Cessna 310H, but also capable of providing the student with realistic experience in the techniques and problems unique to airborne data acquisition. Both the overall system and individual system components were evaluated with respect to several specific parameters. While this section discusses the overall system requirements, specifications peculiar to each system component are discussed in following sections of this report.

Due to limitations in aircraft capability, hardware availability, and financial resources, the design of the complete instrumentation system had to be carefully evaluated with respect to the criteria:

1) Number and complexity of sensor inputs
2) Type of measurement readout
3) Value of system outputs as both parameter readouts and as instructional tools
4) Desired accuracy and dynamic response of the system
5) Availability of hardware and finances
6) Aircraft restrictions

Because the primary purpose of the aircraft was to develop an airborne method of instruction in the study of flight

evaluation techniques, the sensor inputs must be compatible with those discussed in the course texts. A thorough study of the Naval Air Test Center Fixed Wing Stability and Control Manual and the NATC Performance Testing Manual disclosed that a system incorporating inputs of control forces, control positions, normal acceleration, angles of attack and yaw, and accurate indications of airspeed and altitude, provided sufficient data inputs to operate the aircraft as a flying classroom coincidental with material covered in the texts. Additional inputs of roll, pitch, and yaw rates, and total temperature would enable the Cessna 310H to perform all tests described in the stability and control manual. Design of such a system with regard to previously discussed constraints was the fundamental goal of this project.

The efficient utilization of this aircraft as a flying classroom prescribed a data readout system that provided simultaneous outputs by means of simply constructed, easily operated devices. Although accuracy was desired, the fact that the system was initially to be used to study procedures and techniques, rather than obtain original scientific measurements, removed the constraint of producing highly accurate dynamic response readouts. Since future utilization of the aircraft will require such measurements, a means of incorporating sophisticated recording devices with the present sensors must be considered.

A normal test flight crew will consist of a pilot and three students. The pilot will not be involved in parameter

measurement, so the system must not in any way detract from his flying capability. The student crew will normally change with each flight. Therefore, the system must be simple enough to ensure proper calibration and operation with minimum prior training.

Financial limitations required the use of hardware obtained from various government facilities. Fortunately, access to a number of installations conducting flight evaluation studies has ensured the availability of adequate, high quality instruments at relatively little cost. In most cases installation of instrumentation will be accomplished by technicians assigned to the Aeronautics Department.

Use of the Cessna 310H as an instrument platform instead of a heavier, more powerful military aircraft required that careful consideration be given to aircraft capabilities and limitations. Operation of the four-seat, instead of the normal six-seat, configuration practically removes the possibility of exceeding the aircraft's maximum gross weight restriction of 5100 pounds. However, the effect of the instrumentation system on other aircraft structural and electrical limitations had to be considered. These include center of gravity excursions, structural configuration, electrical power, and flight envelope considerations. Applicable Federal Aviation Administration regulations pertaining to aircraft modifications must be consulted due to the aircraft's civilian registry.

The operation of a civilian registered aircraft by a military activity placed several constraints on the project.

Because the aircraft is not in custody of the Navy, many portions of OPNAV 3710.7 series of instructions covering Naval aircraft operation do not apply, and FAA regulations must be followed. However, since in this case the operating activity is military, certain procedures applicable to a military operation must also be complied with. A complete operations manual covering all aspects of this unique situation was written, and a laboratory procedures manual prepared. Modification and operation of the Aeronautics Department's Cessna 310H must be in accordance with the guidelines expressed therein.

IV. AIRCRAFT

The Cessna 310H is a six-place, low wing, twin engine monoplane with fully retractable landing gear. Built in 1963, the basic purpose of the aircraft is personnel transportation, although it can be used for transportation of light cargo in addition to other liaison missions. The Postgraduate School's 310H is operated in a four-place configuration, with the rear seats removed to provide space for part of the instrumentation system. A photograph and diagrams of the overall aircraft are shown in Figures 1, 2 and 3.

The aircraft is powered by two horizontally opposed, six-cylinder, Continental Model IO-470-D engines rated at 260 horsepower at 2625 RPM. Engine power is controlled by manual operation of throttles, mixture controls, and propeller RPM.

Fuel is supplied by two 51-gallon main tip tanks and two 15-gallon auxiliary tanks located in the wings. Depending on conditions and power settings, the 310H can remain airborne over eight hours if necessary. Maximum range is approximately 1250 miles.

Electrical power is supplied by a 28-volt direct current system. No alternating current power is available. A 50-ampere generator is mounted on each engine, and two 12-volt batteries in series supply energy to the system when the generators are inoperative. An external power receptacle mounted in the left wing allows the use of external power for

extended ground operation of electrical equipment or for starting when battery power is low. All circuits are protected by "push to reset" type circuit breakers.

Although dual controls are provided, the instrument panel contains only one set of flight instruments and one set of engine instruments. Use of any of the normal flight instruments for accurate airborne parameter measurement is considered marginal.

The available flight envelope and flying characteristics of the Cessna 310H make it ideal for use as an airborne classroom. Maximum gross weight is 5100 pounds, with a useful load capability of 1805 pounds before fueling. Fully loaded stall speed is 75 MPH IAS with landing gear down and flaps at 45 degrees. Maximum cruising speed is 210 MPH IAS, with a 'never exceed' limitation of 254 MPH. Maximum gross weight rate of climb is 1690 feet per minute for two engine operation, and single engine flight characteristics are quite satisfactory. An oxygen system allows operation, if desired, to 20,000 feet.

The aircraft is very susceptible to gust and turbulence disturbances, particularly about the yaw axis. When the landing gear and the flaps are down, a strong dutch roll mode is evident. Attainment of a steady state flight condition in the dirty configuration is very difficult.

In general, the flight characteristics of the 310H, although acceptable throughout the flight envelope, will provide valuable experience in solving the problems involved in acquiring satisfactory airborne data.

Detailed information concerning the operation and the capabilities of the Cessna 310H is available in several of the sources listed in the bibliography.

V. INSTRUMENTATION

A. STUDENT INSTRUMENT CONSOLES

The primary purpose of the instrumentation system installed in the Naval Postgraduate School Cessna 310H is to provide an efficient means of airborne data acquisition for students enrolled in the flight evaluation techniques two-course sequence. Several data readout methods were considered, including oscillographs, tape recorders, photo panels, and visual displays. Constraints of space, weight, student utilization, and real time requirements were the factors that influenced the decision to use individual student consoles with manual recording of data. Although accuracy and dynamic response measurements of such a system are inferior to other methods, the individual consoles are far superior to the other readout systems as instructional tools. With three student instrument consoles installed, the Cessna 310H can effectively operate as a flying classroom.

The Cessna 310H was originally designed as a six-place aircraft. However, to provide space for support instrumentation and to allow loading flexibility, the two rear seats were removed. Since the pilot must not be burdened with parameter measurements, only three instrument consoles were designed and installed. The co-pilot's console is located between the pilot's and co-pilot's seats as shown in Figure 5. The consoles for

the rear seat passengers are secured by brackets to the back of the seat immediately forward of their respective stations (Figure 6).

Inputs to the consoles consists of both electrical signals and differential pressures. The nine electrical sensors are:

1) Angle of attack
2) Sideslip angle
3) Elevator force
4) Aileron force
5) Rudder force
6) Elevator position
7) Aileron position
8) Rudder position
9) Normal acceleration

Three additional pitot-static systems were installed: one in the wing tip mounted boom, and one under each wing approximately four feet outboard of the propeller arc. In addition, a 150-foot trailing cone system provided an alternate static source. Connection of the various pitot-static systems to the instrument consoles was straightforward. The boom-mounted system was connected to the co-pilot's console, and the wing-mounted systems were connected to the respective rear seat consoles. The trailing cone static tube can be connected to any station.

Each student instrument console contains a standard triple-pointer altimeter, an airspeed indicator, a digital voltmeter, and a five-position 2-pole rotary switch. The left rear console also contains an elapsed time clock.

The airspeed indicators used were made by Kollsman Instrument Corporation and are shown in Figure 7. The indicators, which were calibrated in knots, have a needle for the outer scale providing a rough indication of airspeed between 100 knot scribes, and a cylindrical drum graduated in two-know increments from zero to 100 knots rotating horizontally and located just above center on the face of the indicator. To read airspeed, observe the position of the 100-know pointer, and then add to the lower hundred indication the number below the arrow in the window of the circular drum. All of the airspeed indicators and altimeters installed in the consoles were obtained on loan from the U. S. Air Force flight instrumentation pool, Edwards Air Force Base, California.

The display system for the output signals from the nine electrical sensors is shown schematically in Figure 8. A system was required that allowed great flexibility in parameter measurement selection but was also relatively simple to install and operate. The system installed in N164X uses a J-Box for sensor output collection, wire plug-in jacks for flexibility in patching signals to individual consoles, a five-position 2-pole rotary switch in each console, and a DATEL DM-1000 digital voltmeter in each console.

The J-Box, shown in Figure 9 and Figure 10, is an aluminum chassis box 8" x 17" x 3" that has been machined to allow mounting of various electrical hardware. Input signals to the J-Box are of two kinds: (1) force and normal acceleration measurements from wheatstone bridge circuitry and amplified by Grant

operational amplifiers and (2) control position and aircraft angles of attack and sideslip measurements obtained from potentiometer voltage changes. Figures 11 through 15 are diagrams of the input circuitry.

The eight wires relaying the four input signals from the amplifiers are connected directly to the J-Box by a 10-wire Bendix cannon plug. Each amplified output signal is then wired to one of 15 pairs of jack plugs mounted on the top face of the J-Box. Although a total of only nine signals are currently being processed, 15 jack plugs were installed to accommodate future system expansion. The jack plugs are color coded red for positive potential and black for negative potential.

All potentiometer circuits contain balance pots that allow voltage nulling for any desired zero settings of the sensor potentiometers. Thus, input signals from these circuits are actually the voltages measured between the sliding arm contacts of the sensor and balance potentiometers. This is shown in Figure 11 and Figure 14. Because the voltage is measured between two resistors and not to a common circuit ground, a floating ground system must be utilized to ensure proper voltage readout. The reference voltage for each individual circuit is unique; hence the wiring must be designed so that each individual voltage reference is available at the readout devices. To accomplish this, the balance pots were physically located within the J-Box. The previously discussed reference voltages are those voltages at the sliding arm contacts of the balance

pots measured relative to ground. They are considered the negative potentials of the potentiometer circuits and are connected to individual black jack-plugs. The corresponding positive potentials are obtained from the sliding arm contacts of the sensor potentiometers and are fed to the J-Box by a Bendix five-wire cannon plug. They are then wired to the appropriate red jack-plug.

A study of the measurements required for any test flight described in the Navy Test Pilot School Fixed Wing Stability and Control Manual showed that a maximum of five electrical readouts were required for any given test. Therefore, the system installed in N164X enables each of the three students to observe any of five parameters simply by rotating his console switch. As stated before, a total of nine electrical sensor outputs are wired into the J-Box. Prior to, or even during the flight, the test coordinator decides which five electrical parameters are needed. He then patches the required pairs of input jacks to the five available output jack pairs located in the J-Box. The rotary switches of the three consoles are each wired directly to the five pairs of output jacks (Figure 10). The various output readouts can be changed simply by connecting the output jack leads to different input jacks.

The five-position console rotary switches are wired to DATEL model DM 1000 digital voltmeters. A photo of the DM 1000 is shown in Figure 16, and the specifications are shown in Figure 17. The maximum display signal of the DM 1000 is

$\pm$ 1.999 volts. Because of this constraint, and certain peculiarities of the system strain gage amplifiers, it was impractical to utilize a consistent readout scaling factor; i.e., one pound force or one degree deflection could not always equal a fixed millivolt output for every circuit in the system. It was therefore decided to calibrate each electrical sensor circuit to take maximum advantage of the characteristics of the amplifier and voltmeter in that particular circuit and to utilize calibration tables to convert the voltage readings to meaningful data. The calibration tables for all electrical sensors are included as Appendix B, and a copy is available at each student console. It should be emphasized that in most cases the scaling factors differ among the electrical outputs, and the tables must be consulted to obtain correct sensor values.

B. VANE SYSTEM

A boom that extends ahead of the aircraft into the free airstream allows accurate measurement of certain parameters without influence of the aircraft mass on the airflow. Usually, such a boom is mounted on the nose of the aircraft, but structural design of the Cessna 310H, however, prohibited such an installation on N164X without major structural modification to the aircraft's forward fuselage area. Instead, the boom was mounted under the left wing immediately below the junction of the wing spars with the left tip tank. Since the tip tanks are secured to the spars by means of 5/16 inch AN STD aircraft bolts, properly designed boom support brackets can be attached

to the spars simply by replacing the original aircraft bolts by longer bolts of the same diameter. To validate design and installation, blueprints for a similar boom and associated hardware were obtained from the flight test branch, Wallace Division, Cessna Aircraft Company, Wichita, Kansas. A photograph of the overall boom assembly is shown in Figure 18.

The boom itself was constructed of a two-inch diameter, 0.125 inch wall thickness, 1024 seamless tube, ninety inches long. Two adapters that enable the boom to be bolted to the spar brackets were welded to the tube as shown in Figure 19. In addition a circular fitting that attaches the tip tank support braces was fastened to the boom as shown in Figure 20. The wing spar brackets (Figure 21) were machined from one-quarter-inch 2014T6 aluminum to Cessna's specifications. The tip tank support braces were added to ensure dynamic stability at high speed, and were fabricated of 1/8-inch sheet aluminum and bent into the proper shape. They are bolted to the boom clamp and each secured to the forward part of the tip tank by two 8/32 machine screws. The screws fit into holes previously drilled into the tank structure to accommodate the forward tank fairing.

All the brackets and braces were designed to allow relatively easy removal of the boom assembly from the aircraft. To remove the boom, loosen the two bolts attaching the boom to the wing spar brackets, unscrew the four tip tank screws, and disconnect electrical and pitot-static lines by using the quick-disconnects located near the base of the boom. If the

aircraft is to be flown without the boom, the exposed electrical and pitot-static lines must be stored within the bottom tip tank fairing.

Parameter sensors were mounted on the boom by means of a yaw and pitch sensor (YAPS) head that was obtained from the Army Flight Test Facility, Edwards Air Force Base, California. This particular device electrically measures yaw and pitch angles and also provides a self-aligning pitot-static source in the free airstream. Photographs of the boom head are shown in Figures 22 and 23.

The pitot-static tube in the nose of the assembly can align with the free stream due to the flexible rubber coupling that connects it to the vane housing. Airflow striking the cylindrical fairing mounted aft of the pitot-static ports creates a moment at the flexible coupling that aligns the nose of the tube parallel to the airflow. Two one-eighth inch plastic tubes extend from the pitot-static ports through the rubber coupling and vane housing aft through the boom to the previously discussed quick-connects. The boom-mounted pitot-static tube system is further discussed in a later section of this report.

Two identical light-weight aluminum vanes, mounted perpendicular to each other on the vane housing, deflect with the relative wind and provide yaw and angle of attack measurements. The vanes are connected to 5-K ohm potentiometers enclosed within the boom head so that vane deflection changes the relative circuit resistance, thus changing the measured

output voltage. Direct calibration of the potentiometers by adjusting the resistance to correspond to a given vane deflection proved highly impractical since the boom head had to be completely disassembled in order to adjust the pot position. Instead, 5-K ohm balance potentiometers were introduced in the circuits parallel to the sensor pots. The vane output reading was taken as the voltage developed between the sliding arm of the sensor pot and the fixed reference voltage at the sliding arm of the balance pot.

Each vane output was calibrated by mounting the boom head in the Naval Postgraduate School low speed wind tunnel so the vane housing tube axis was parallel to the airflow. The housing cylinder was then rotated until the support arm of the vane being calibrated was horizontal. The airflow then aligned the vane parallel to the housing cylinder, corresponding to a zero angle of attack or yaw reading. The appropriate balance potentiometer, located in the system J-Box, was then adjusted to null any voltage that existed between the sliding arm contacts of the two pots. It should be noted that because of the incorporation of a floating ground system, any desired vane deflection may be used as the zero reference simply by readjusting the balance potentiometer to null the voltage produced by the desired vane deflection.

Calibration continued by inclining the housing assembly at known angles of attack, aligning the vane with the airflow, and recording the corresponding voltage. The second vane was calibrated in a similar fashion after rotating the housing

cylinder through ninety degrees to align the vane arm horizontally in the wind tunnel. A photograph of the boom head being calibrated is included as Figure 24.

Based on previous experience, the decision was made to calibrate the yaw vane to plus and minus thirty degrees deflection, and the angle of attack vane to plus and minus twenty degrees. Although this range is somewhat excessive, the voltage outputs corresponding to these full scale deflections are compatible with the readout system limitations. Calibration output voltages are tabulated in Tables I and II, Appendix B, and calibration curves are shown in Figures 25 and 26.

The output voltage for the desired full scale vane deflections using the 5-K ohm potentiometers depends on both the voltage across the pot and on the wire-wound resistor sensitivity built into the pot. Since the latter was not controllable, the voltage output corresponding to full scale deflection was kept within range of the console voltmeter by dropping the 28 volt D.C. aircraft voltage to 15 volts through the use of National Semiconductor miniaturized voltage regulators shown in Figure 27. The voltage regulators were mounted in the J-Box. They not only kept the output voltage within limits but also supplied a steady fifteen volts across the potentiometers regardless of aircraft power fluctuations.

The boom head assembly was semi-permanently mounted on the boom by means of safety wired machine screws. Prior to drilling the necessary holes, the boom head was rotated until the

forward vane was sighted vertical and the aft vane horizontal. The YAPS head was installed after the boom tube was mounted on the aircraft.

C. FLIGHT CONTROL SYSTEM

The flight control system of the Cessna 310H is entirely mechanical. The primary control surfaces (ailerons, rudder and elevator) are positioned by dual control wheels and rudder pedals through a conventional cable system. The aircraft can be trimmed about all three axes with trim wheels located on the engine control pedestal. Positioning of trim tabs is accomplished through a conventional cable and pulley system to screw jacks and push-pull tubes which activate the tabs. Diagrams of the aircraft flight control systems are shown in Figures 28, 29, and 30.

1. Measurement of Control Forces

Strain gages incorporated in wheatstone bridge circuits provide the most efficient method of control force measurement. Proper positioning and mounting of the strain gages is essential in order to ensure correct correlation between forces applied to the cockpit controls and output voltage changes measured across the respective wheatstone bridge.

Because the voltage changes associated with strain gages are so small, signal amplification was deemed necessary. This was accomplished by means of Grant model DCA8-3 operational amplifiers. (Figure 31).

Advantages of this type of amplifier are:

1) Small size (2 in. x 2 3/15 in. x 1 in.)

2) Simplified input circuitry

3) High gain

4) Low drift

5) Extremely rugged

6) Ease of access for zero adjustment and gain adjustment

7) Compatible with aircraft 28 volt direct current electrical system

8) Insensitivity to input power fluctuations

9) Internally produced excitation voltage

The only two negative aspects of amplifier operation are extreme sensitivity of the zero adjustment, and reversed polarity output saturation. Although the Grant op-amp is capable of producing a 5-volt positive polarity output, the amplifier saturates at approximately 0.695 volts when inputs from the bridge circuits reverse polarity. Means of overcoming this difficulty are discussed in individual cases where it is appropriate.

The Grant op-amps utilize solid state circuitry with simplified external wiring. Two six-wire cannon plugs allow expeditious amplifier hook up. One cannon plug connects the four nodes of the bridge circuit. Five-volt excitation voltage is applied from the A and D connectors and the unamplified signal is received through connector B and C. Connectors E and F provide case grounding. The second cannon plug receives

the 28-volt power through connectors B and C, and the amplified signal output is obtained through connectors D and E. In this plug, connectors F and A are ground.

The Grant model DCA8-3 amplifier incorporates adjustable zero and gain settings. These adjustments are made by inserting a small jeweler's screwdriver in the appropriate slot near the cannon plugs. Normally the gain of each amplifier will be set during initial calibration and will not be readjusted except for recalibration. The zero adjustment, however, is extremely sensitive and tends to drift slightly. The zero will be set prior to, and when necessary during, each flight.

The small amplitudes of the strain gage signals make the control force measuring system very susceptible to electrical noise. In order to preserve signal integrity, the operational amplifiers should be located as close to the bridge circuits as possible. In the Cessna 310H, however, space limitations necessitated all Grant op-amps being contained in a chassis box mounted aft of the passenger seats. This allows easy access for zero adjustment. All wiring between the individual bridge circuits and the amplifiers utilizes shielded cable to reduce electrical noise.

a. Aileron and Elevator Forces

Aileron and elevator forces are measured by strain gages mounted on a specially designed control wheel which replaced the standard co-pilot's wheel as shown in Figure 32. Blueprints for the wheel were obtained from the Wallace

Division of Cessna Aircraft Company, Wichita, Kansas. The wheel, which was completely fabricated by the Aeronautics Department machine shop, is identical to those used by the Cessna Aircraft Company in its flight test programs.

The strain-gaged control wheel (Figures 33 and 34) is basically two cantilever beams connected by a pipe adapter, with a hand grip mounted on each end. Thin aluminum plates are positioned to provide protection for the strain gages and associated wiring. The cantilever beams are rectangular in cross-section so that, when properly mounted, applied aileron and elevator forces are normal to the respective cantilever surfaces. This allows independent measurement of aileron and elevator forces.

Each force system uses four type C-9-171 strain gages wired to form a 350-ohm wheatstone bridge. The bridge circuits are connected to the Grant operational amplifiers in the manner previously discussed.

Initial calibration of the aileron and elevator force systems was extensive. In all calibrations involving the Grant operational amplifiers the amplified signal was calibrated. For aileron calibration the control wheel was mounted so that a moment could be applied with calibrated weights through a cable-pulley arrangement. Because of the previously discussed reverse polarity saturation, full scale output must be less than 0.695 volts in order to obtain accurate force indications in both left wing down and right wing down directions. Since maximum required aileron force

is 40 pounds, a scale of 15 millivolts per pound was chosen. This gives a theoretical output of $\pm$ 600 millivolts for a 40-pound aileron force applied at the center of the vertical hand grip bar. Calibration was achieved by adjusting the gain and zero settings to produce proper voltage output for a known moment. Because the output was slightly non-linear, it was decided to adjust the gain to be as linear as possible in the lower force range where most measurements will be taken. Therefore as can be seen from Table III of the tabulated calibration data, the linearized scale factor of 15 millivolts per pound cannot be used above 7 pounds. This is not considered a problem since, as discussed previously, use of a single meter for multiple parameter output prohibits the reasonable use of one-to-one correlation between voltage output and actual parameter values. The actual parameter value must be determined by looking up the output voltage in a calibration table.

Elevator force calibration was conducted in a manner similar to the aileron force calibration. Because the elevator force limits ranged from 70 pounds nose up to 40 pounds nose down, reverse polarity saturation was avoided by adjusting the amplified signal output for approximately 15 millivolts per pound. A known force was applied in both directions at the vertical handgrips on the control wheel and the gain and zero settings adjusted to provide the desired voltage output. As in the case of the aileron calibration, slight non-linearities in the system necessitated adjustment of the amplifier to

provide linear low range response (Table IV). Calibration curves for the aileron and elevator force sensors are shown in Figures 35 and 36.

b. Rudder Forces

The rudder force sensing system consisted of a transducer on each rudder pedal on the co-pilot's side with the output from each transducer fed into an operational amplifier. The sign of the signal from the right rudder pedal transducer is changed by reversing the leads coming out of the operational amplifier for that rudder pedal. This then makes the signal from the left rudder pedal positive and the signal from the right rudder pedal negative. The signals are then fed into an electrical algebraic summer (Figure 37). The signal out of the summer is then fed into the J-Box for patching to the appropriate console.

The rudder pedal transducers were made by Radiation Incorporated of Los Angeles (Figure 38). The transducers were installed on the co-pilot's rudder pedals as depicted in Figure 39. Serial Number 43 was used for the right rudder pedal and Serial Number 105 for the left. A calibration curve for the two transducers is shown in Figure 40. The curve is linear throughout the range zero to 150 pounds.

The electrical signal is developed in a 350-ohm wheatstone bridge made up of strain gages (Figure 13). The signal is then fed to the operational amplifier. The output of the amplifier was adjusted to approximately 10 millivolts

of amplified signal output per pound of rudder pedal force (Table V).

The sign of the electrical signal for the right rudder pedal is reversed so that the signal out of the algebraic summer is the differential signal between the two rudder pedals. This is then the differential force between the two rudder pedals when the pilot has both feet on the pedals during a maneuver. Since the range of the digital voltmeters is $\pm$ 1.999 volts and the range of the rudder pedal force signal is $\pm$ 1.500 volts, no further processing of the signal was required.

2. Measurement of Control Surface Positions

The control surfaces of the Cessna 310H are positioned by the pilot or co-pilot through mechanical linkages of cables, pulleys, and bell-cranks, as shown in Figures 28 through 30. Since direct movement of the linkage system is required for any control deflection, determination of control surface positions can be accomplished by measuring the displacements of the linkage assemblies, preferably at points near the control surfaces in order to minimize error due to cable elongation.

After several means of displacement measurement were investigated, it was decided to utilize the method employed by N.A.S.A. Flight Test Center, Edwards Air Force Base, California, in their flight evaluation projects. Three linear displacement transducers were purchased from Space-Age Control, Inc. of Palmdale, California. This transducer, shown in

Figure 41, consists of a 2000-ohm rotating-arm potentiometer enclosed in an aluminum housing. The rotating shaft of the sliding contact is connected to a spring-tensioned two and one-half turn spool enclosed within a drum mounted on the side of the potentiometer. A flexible wire cable is wound on the spool and extends through an opening in the drum. This particular model was designed so that as the cable is unwound from the spool to a maximum of six inches, slide contact resistance varies from zero to two thousand ohms, the full resistance of the pot.

Direct observation showed that the maximum movement of any of the control linkages from the neutral position was plus or minus 2 3/8 inches, a value well within the useful transducer range of plus or minus three inches. The installation of the transducers, although simple in concept, proved difficult because of space and control linkage access limitations, particularly for the aileron installation.

The first step in the installation procedure was to physically lock the control surface in the neutral, or zero deflection position. The transducer was attached to an appropriately designed mounting bracket and then bolted into a suitable position near the control surface bell-crank. A swivel fitting was fastened to the bell-crank to attach the transducer cable. The cable was then unwound from the spool and fastened to the bell-crank so that zero control surface deflection unwound three inches of cable. Control movement either caused more cable to unwind, thus increasing slide

contact resistance, or else, because of the cable retraction force created by the spool spring, caused retraction of the cable, thus decreasing the slide contact resistance.

A shear pin was used to connect the transducer cables to the control bell-crank attachment swivel fittings. The maximum load limit of the shear pin is five pounds. If the potentiometer or cable spool should jam, the transducer cable will detach from the bell-crank when the shear limit is exceeded, thus freeing the control linkage.

The electrical circuitry of the control position measuring system is shown in Figure 14 and is similar to that of the boom-mounted vanes discussed earlier. Five-thousand-ohm balance potentiometers mounted in parallel with the linear displacement transducers allow voltage nulling for neutral control positions. Again, the voltage readout is between the sliding arm contacts of the transducer and balance potentiometers. Because the resistance of the transducer pots is only 2000 ohms, the aircraft 28 volt electrical supply had to be reduced to five volts to prevent the measured output voltages from exceeding the 1.999 volt Datel Digital Voltmeter limitation. This was accomplished through the use of National Semiconductor miniature voltage regulators shown in Figure 27. The voltage regulators were mounted within the J-Box near the position circuit balance potentiometers.

The elevator and rudder position transducers were mounted in the rear fuselage section of the aircraft near the elevator and rudder hinge lines as shown in Figures 42 and 43.

The aileron transducer was mounted in the left wing immediately inboard of the aileron bell-crank as shown in Figure 44. The maximum control deflections measured by the position transducers are:

Elevator: + 25 degrees, - 15 degrees

Rudder: $\pm$ 25 degrees

Aileron: $\pm$ 20 degrees

Because of the previously discussed limitations for single meter readout of multiple inputs, no convenient correlation factor between control deflection and voltage output was possible. Control positions are determined by looking up the measured output voltage in appropriate tables. Project delays have so far prohibited a complete formulation of calibration tables. However, it has been determined that the voltage output of the control surface position transducers is linear throughout the deflection range of each control surface.

D. PITOT-STATIC SYSTEM

The pitot-static system installed in N164X was designed for maximum flexibility. In addition to the original system in the aircraft, three additional systems were installed that will enable evaluation of error in pitot-static systems due to physical location of the pitot tube on the aircraft. A trailing cone system was also incorporated as an alternate static source that could be used with any of the additional three pitot systems installed.

No changes were made in the original system in the aircraft. The pitot tube for that system is in the nose of the aircraft and is pictured in Figure 45. The pressures sensed are used for the altimeter and airspeed indicator on the pilot and co-pilot's instrument panel.

A head for sensing angle of attack and yaw was installed on the port wing tip (Figure 23). This head also included a pitot-static sensor with a fairing to keep the pitot tube aligned with the relative wind. The head was mounted on a boom which was designed to keep the pitot sensing system far enough ahead of the wing to prevent erroneous readings due to disturbances caused by the wing moving through the air. This system was used as the pitot source for the console installed between the pilot's and co-pilot's seat. Plastic tubing was used in the boom, routed through the wing and up to the engine nacelle. Aluminum tubing was used through the engine nacelle due to temperature considerations (Figure 46). Plastic tubing was again used from the inboard side of the engine nacelle through the fuselage, where a small hole was drilled at station 25 (Figures 47 and 48), then to the console. The static pressure from the head is brought into a selector valve (Figure 49) at the console where the decision can be made to use that pressure or the static pressure from the trailing cone.

Two identical pitot tubes were manufactured by the Aeronautics Department machine shop. One pitot tube was mounted on the underside of each wing approximately four feet

outboard of the propeller tip arc at station 161 (Figures 47, 19, and 50). A special adapter plate was made for each pitot tube so it could be installed in place of the access panel on the respective wing. As in the pitot system for the boom, plastic tubing was used up to the engine nacelle, aluminum tubing through the nacelle, and plastic tubing from there through the fuselage to the console. The tubing for the pitot tube installed under the port wing is routed alongside the tubing for the pitot system from the boom. The tubing for the pitot tube installed under the starboard wing is threaded through the starboard wing in a manner similar to that done in the port wing.

A trailing cone system was installed in the aircraft as an alternate static pressure source. An access panel was removed from the underside of the fuselage at station 200 (Figure 47) and an adapter plate was made for the cone that could be used in place of that panel. One hundred and fifty feet of one-quarter inch inner diameter clear plastic tubing was used with the cone. It is let out and pulled back in manually, and stowed in the cabin as shown in Figure 49. The static pressure sensed by the cone was put through a three-way fitting and routed to the selector valve at each of the three student consoles where it may be interchanged with the normal static source.

E. ACCELEROMETER

A Statham model ASTC-8.0-350 accelerometer was used to measure the aircraft's normal acceleration (Figure 51).

The range of this accelerometer is negative two g's to positive eight g's. Construction is of the unbonded strain gage bridge design which generally has the strain gages attached to a fixed frame and a force summing member (Figure 15). When the force summing member is displaced, the balance of the bridge is changed, providing an electrical output proportional to the magnitude of the applied force. A Grant operational amplifier was used with the accelerometer to give the desired output of 0.5 volts per g.

The dynamic response of the accelerometer was tested to ensure the voltage output per g was nearly constant at the low frequencies. In addition a phase lag evaluation was made, and from that the natural frequency (ω_n) was determined. With the natural frequency and the dynamic response curve determined the damping ratio, zeta (ξ), and the damped natural frequency (ω_D) were calculated.

The dynamic response calibration was conducted on the shaker table in the Aeronautics Department dynamics lab. A Bentley reluctance gage was used as the reference because its output is linear in the range tested. On the first calibration run the frequency range was zero to 600 Hertz. The voltage outputs of the Bentley and Statham were recorded at various frequencies and are listed in Table VI. A second verification run was made in the frequency range zero to 250 Hertz, and these values are also listed in Table VI. For each frequency the number of g's were computed from

$$g = \frac{A}{K_B} \frac{(2\pi f)^2}{386.1}$$

where A is the output of the Bentley in volts RMS, K_B is the slope of the Bentley calibration curve (350 volts per centimeter), and f is the frequency. The millivolts per g for the Statham were then calculated and are also listed in Table 6. A plot was then made of millivolts per g vs. frequency for both calibration runs (Figure 52). It can be seen that the output of the accelerometer changes very little in the frequency range zero to 100 Hertz.

To determine the natural frequency, a phase lag calibration run was made. The results of that run are recorded in Table VI. Since the phase lag of a linear second order system for all damping ratios is 90 degrees at the natural frequency, the natural frequency of the Statham is 213 Hertz. If the ordinate in Figure 52 is normalized so that the curve intersects the vertical axis at 1.0 (call that the g axis), then $g_{max} = \frac{415}{355} = 1.169$. For systems with damping ratio less than 0.707, $g_{max} = \frac{1}{2\xi\sqrt{1-\xi^2}}$. From this the damping ratio is calculated to be 0.49. Since the damped natural frequency is defined by $\omega_D = \omega_n (1 - 2\xi^2)^{1/2}$ in this case ω_D= 154Hz. The natural frequency and the damped natural frequency of the accelerometer are well out of the range of any airframe frequencies, which are usually less than ten Hertz.

This accelerometer was statically calibrated on a centrifuge for a previous project and found to have a linear

output in the range -2 to +6 g's. As a result, the only calibration required was the two-g turnover method to adjust the zero and gain of the Grant operational amplifier for the desired output (volts per g). The transducer is first placed on a level platform with its sensitive axis perpendicular to the earth's gravitational field. The output of the Grant op-amp was then zeroed for this zero-g condition. The accelerometer was then rotated to the positive one-g condition and the op-amp output was adjusted to +0.501 volts. It was then rotated to the negative one-g condition and the Grant was adjusted to -0.500 volts. Since the acceleration limit of the aircraft is +3.8 g's the maximum voltage output is +1.90 volts. This does not exceed the range of the Datel digital voltmeter and no further processing of this signal was required.

VI. CONCLUSIONS AND RECOMMENDATIONS

Due to the lack of time in which to conduct an in-flight evaluation of the data acquisition system, the instrumentation described in this report should be considered as components of a proposed, rather than a fully operational, data collection system. Undoubtedly, some changes in the design of various components will be made during actual in-flight testing.

No attempt has been made to formulate standard pre-flight and normal operating procedures for the installed instrumentation. It is felt that without proper in-flight evaluation of the capabilities of the system, no meaningful procedures can be developed, and any attempt at standardizing the operation would not be worthwhile.

Fortunately, plans to continue the project for at least three years have already been made and additional theses students have been assigned to the instrumentation flight program. The first step in future development of the system must be the thorough flight testing of all the currently installed instrumentation throughout the aircraft flight envelope. Only then will the operational limitations of the system be known, and after necessary modifications the laboratory phase of the flight evaluation sequence can become tailored to coincide with the aircraft's capabilities.

After the present installation becomes operational, thought must be given to refinement and expansion of the system. Because of the previously discussed limitations of the digital voltmeters in measuring the dynamic response characteristics of the aircraft, an automatic data recording device should be installed. Although not as efficient as a magnetic tape recorder, a sixteen-channel oscillograph is available from the Navy Flight Test Instrumentation Pool at NAS Patuxent River.

Additional parameter sensors can easily be added to the system. In this regard, several instruments have been obtained by the Aeronautics Department for future installation in the aircraft. They include:

1) Three single-axis rate gyros
2) Accurate two-axis position gyro
3) Total temperature probe
4) D.C. to A.C. inverter (28VDC to 115V, 400 cycle A.C.)

In designing the instrumentation system for the Cessna 310, a great deal of professional assistance was received from flight test engineers at the Cessna Aircraft Company, Wallace Division, Wichita, Kansas, and the N.A.S.A. Flight Test Center, the U. S. Army Flight Test Facility, and the U. S. Air Force Flight Test Center, all at Edwards Air Force Base, California. It is highly recommended that these engineers be consulted in the event of any future technical problems concerning the project.

Although the results thus far obtained from the aircraft instrumentation program have been minimal, it is felt that the

data acquisition system now installed in N164X will prove to be successful when the program is finally under way.

APPENDIX A

Figures

Figure 1

CESSNA 310H AIRCRAFT N164X

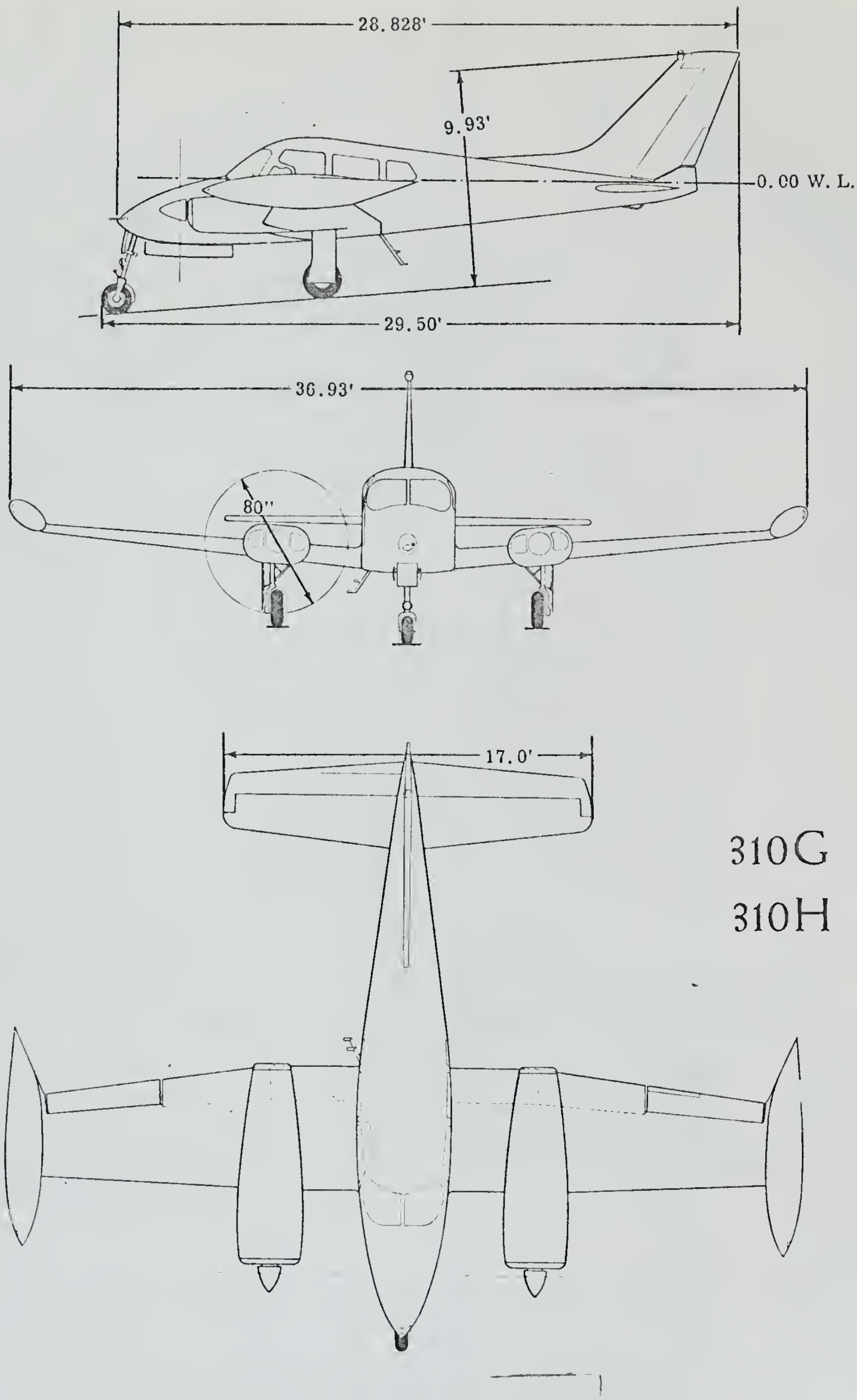
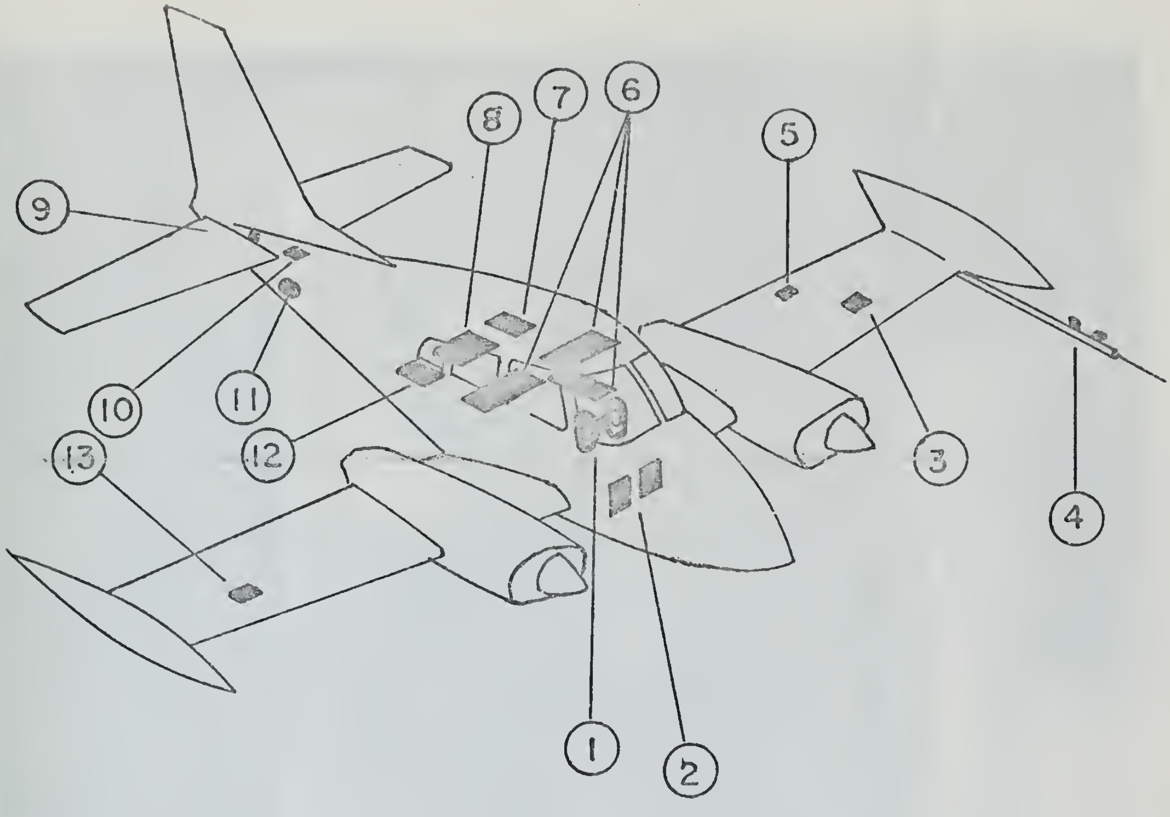


Figure 2
CESSNA 310H DIMENSIONS



1. WHEEL FORCES
2. RUDDER FORCE
3. PORT PITOT TUBE
4. FLIGHT BOOM
5. AILERON POSITION
6. STUDENT CONSOLES
7. ACCELEROMETER
8. J-BOX
9. ELEVATOR POSITION
10. RUDDER POSITION
11. TRAILING CONE
12. AMPLIFIER BOX
13. STBD. PITOT TUBE

Figure 3

DATA ACQUISITION SYSTEM COMPONENTS

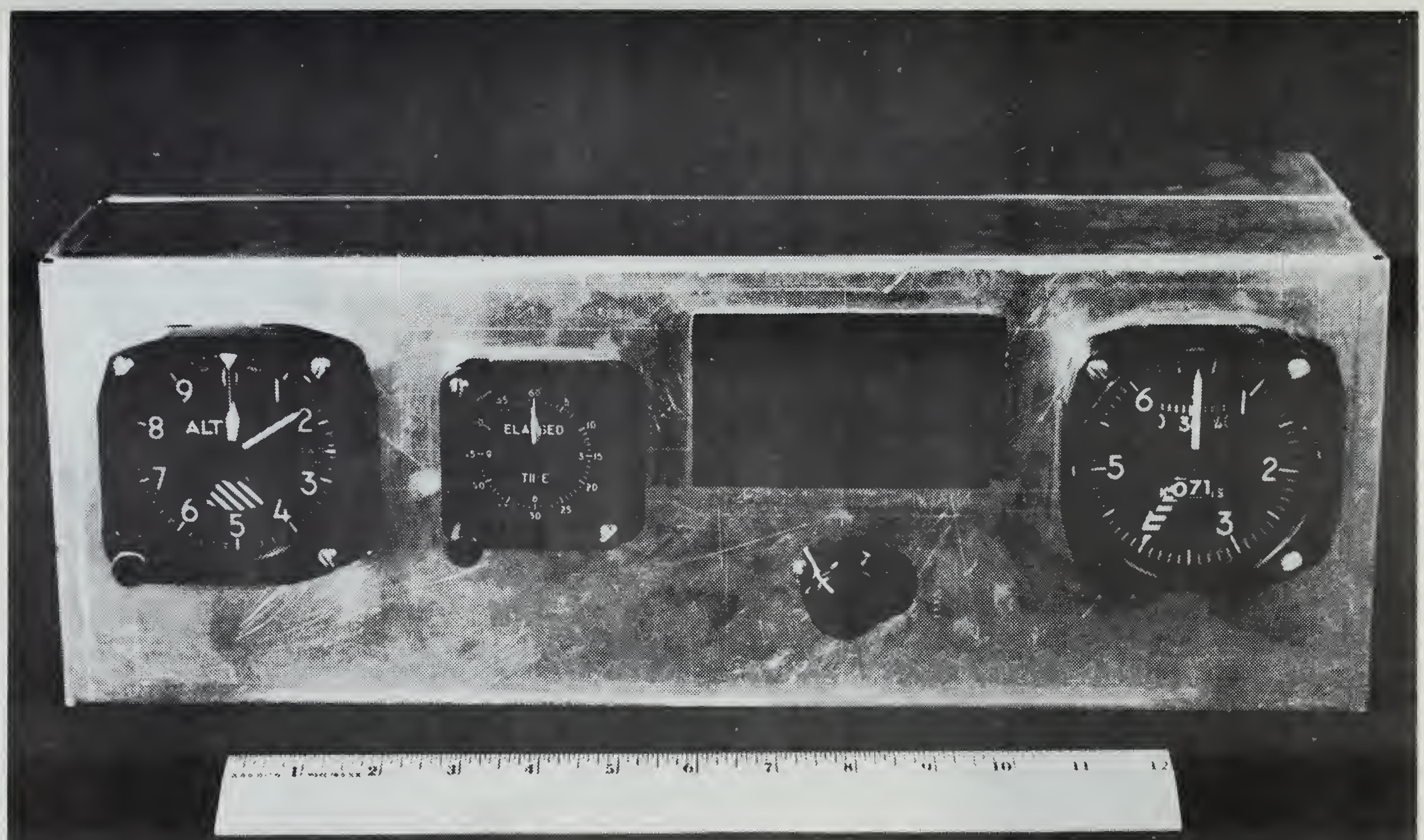


Figure 4

STUDENT INSTRUMENT CONSOLE

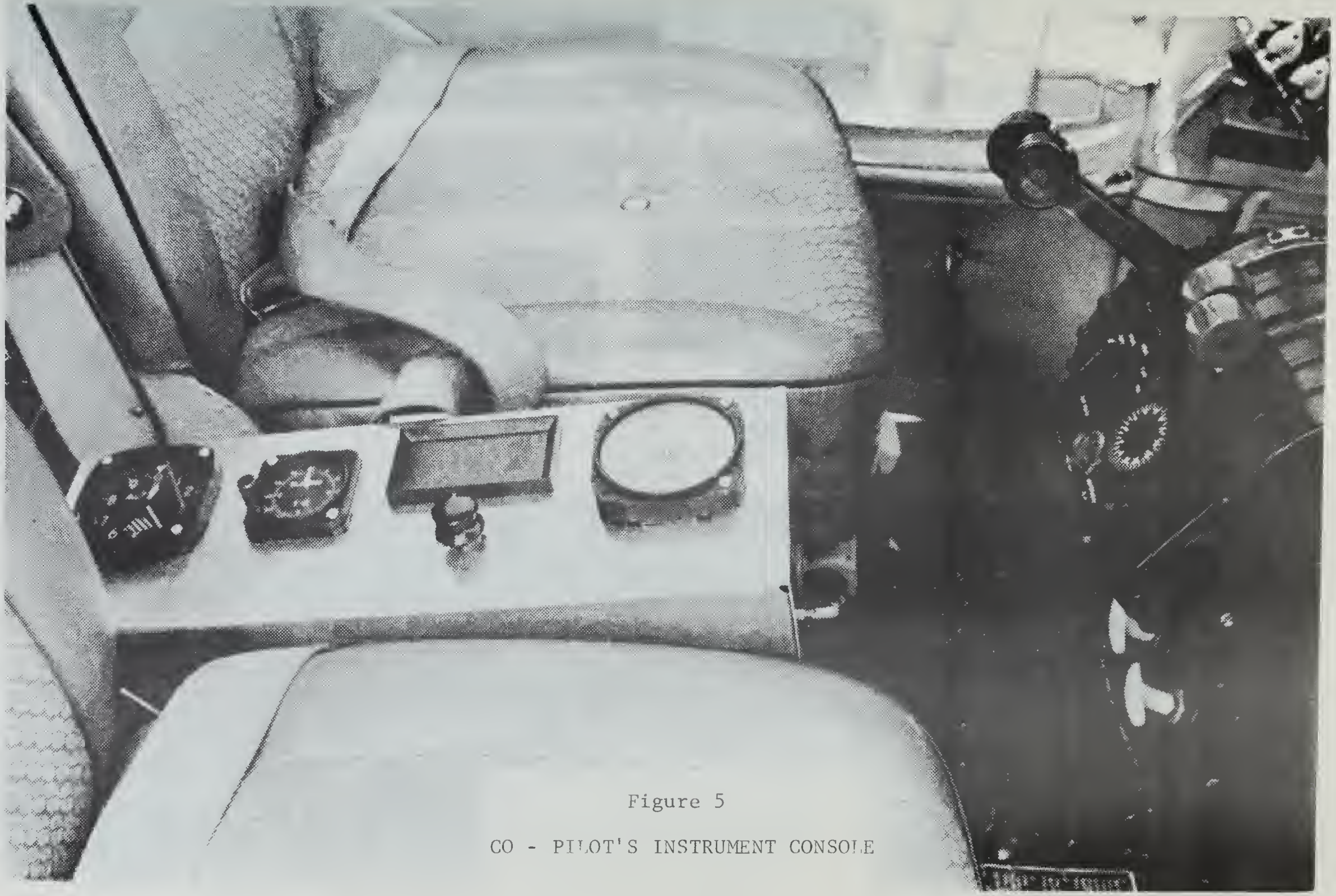


Figure 5

CO - PILOT'S INSTRUMENT CONSOLE

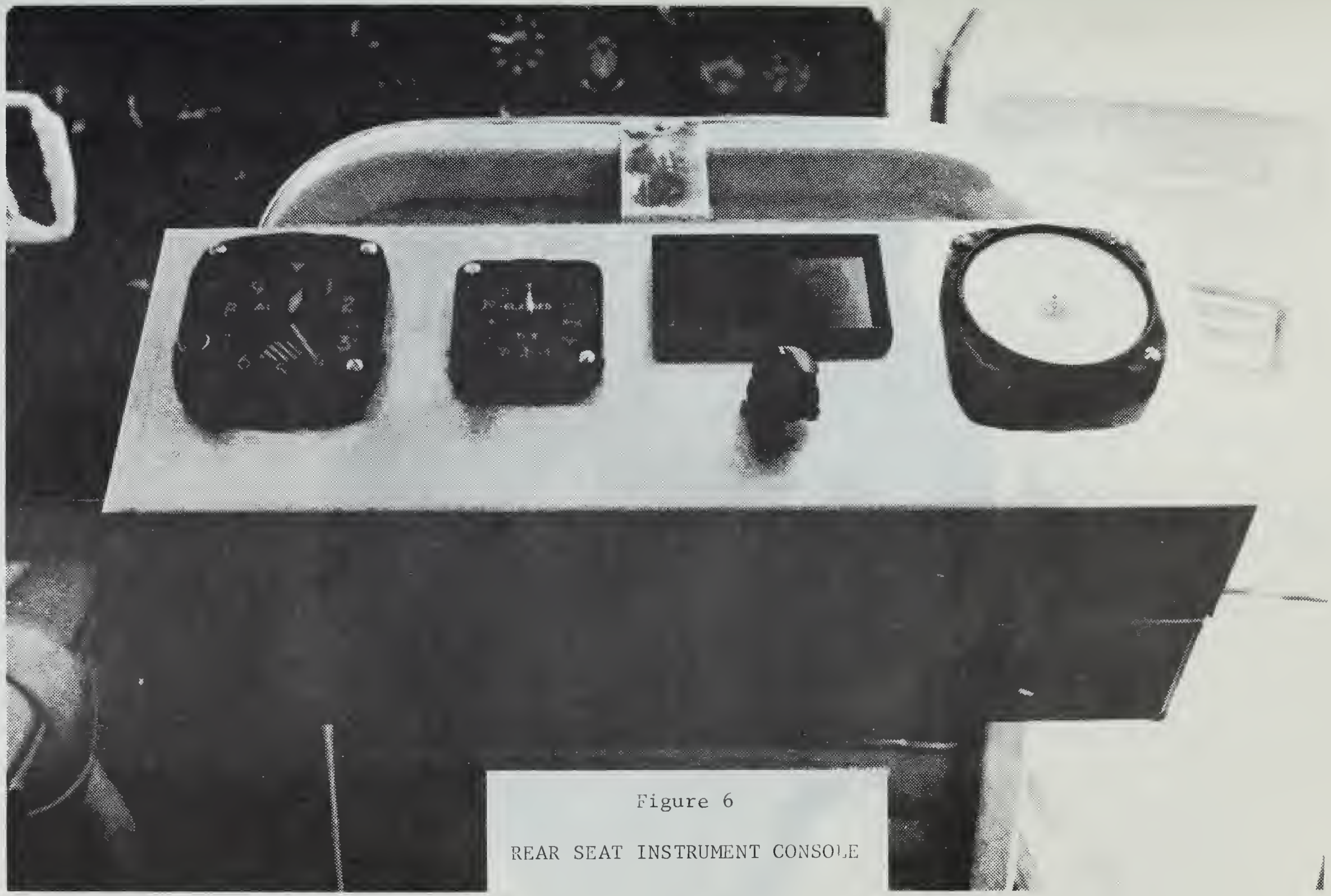


Figure 6

REAR SEAT INSTRUMENT CONSOLE

Figure 7
CONSOLE AIRSPEED INDICATOR

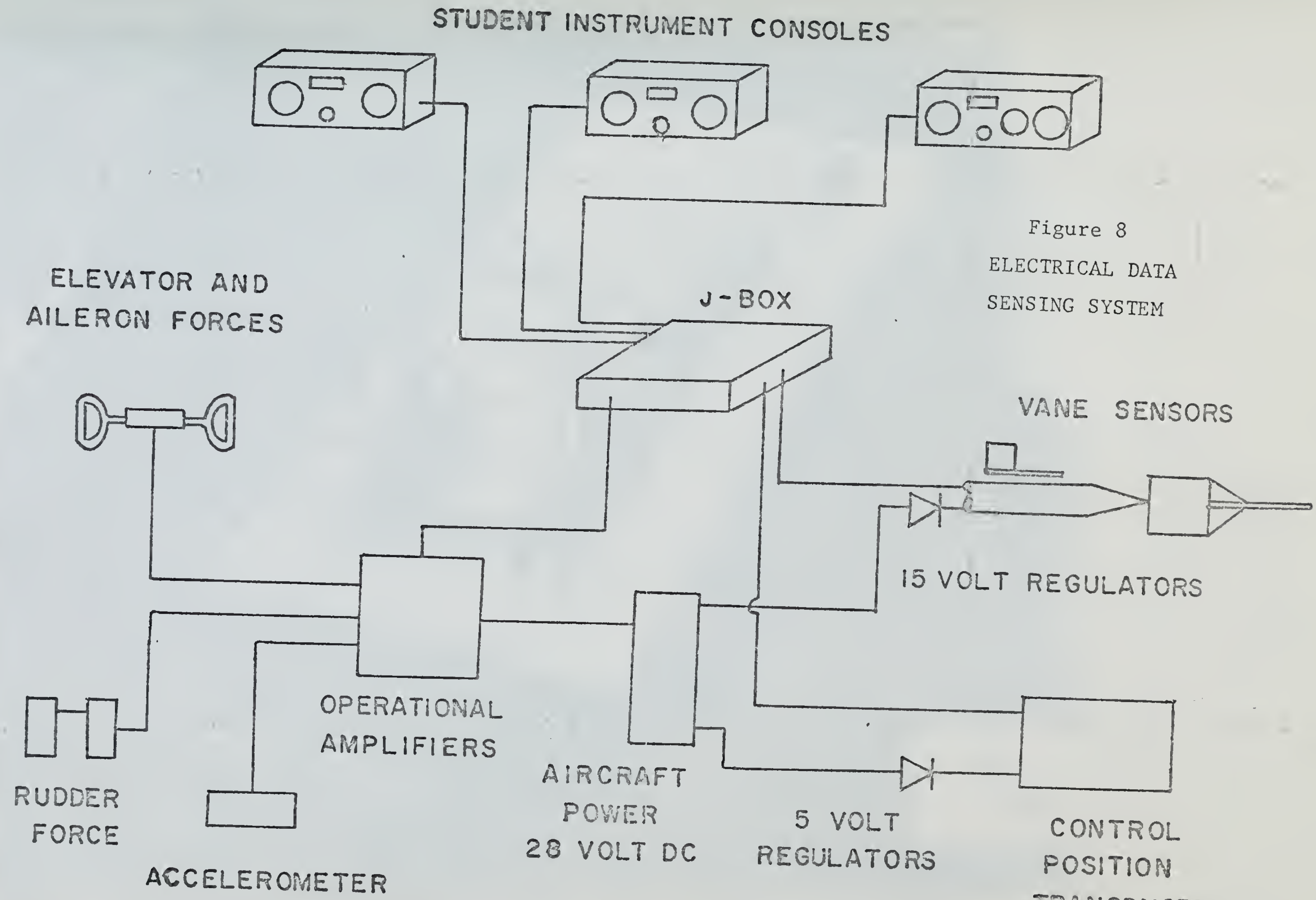


Figure 8
ELECTRICAL DATA
SENSING SYSTEM

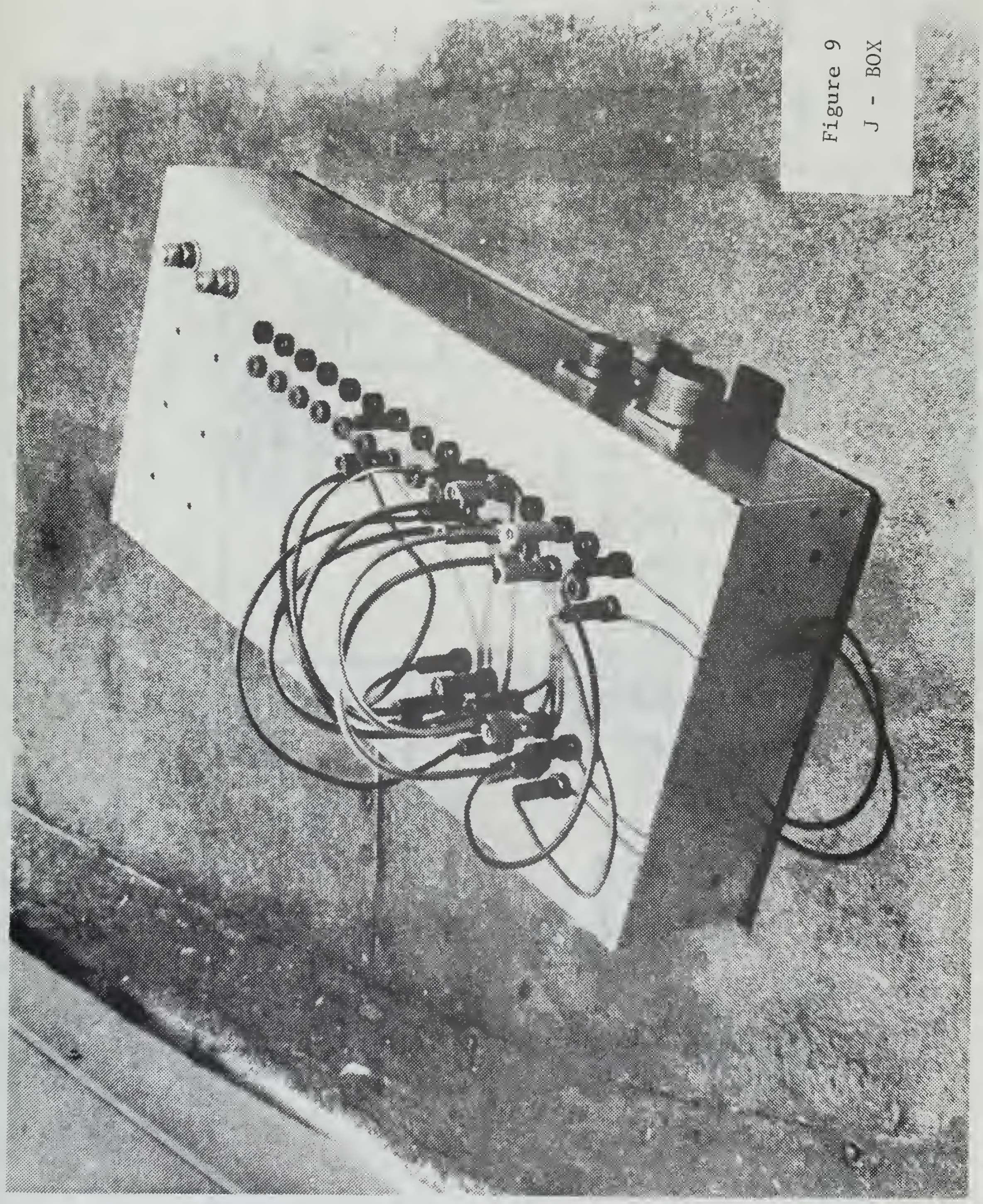


Figure 9
J - BOX

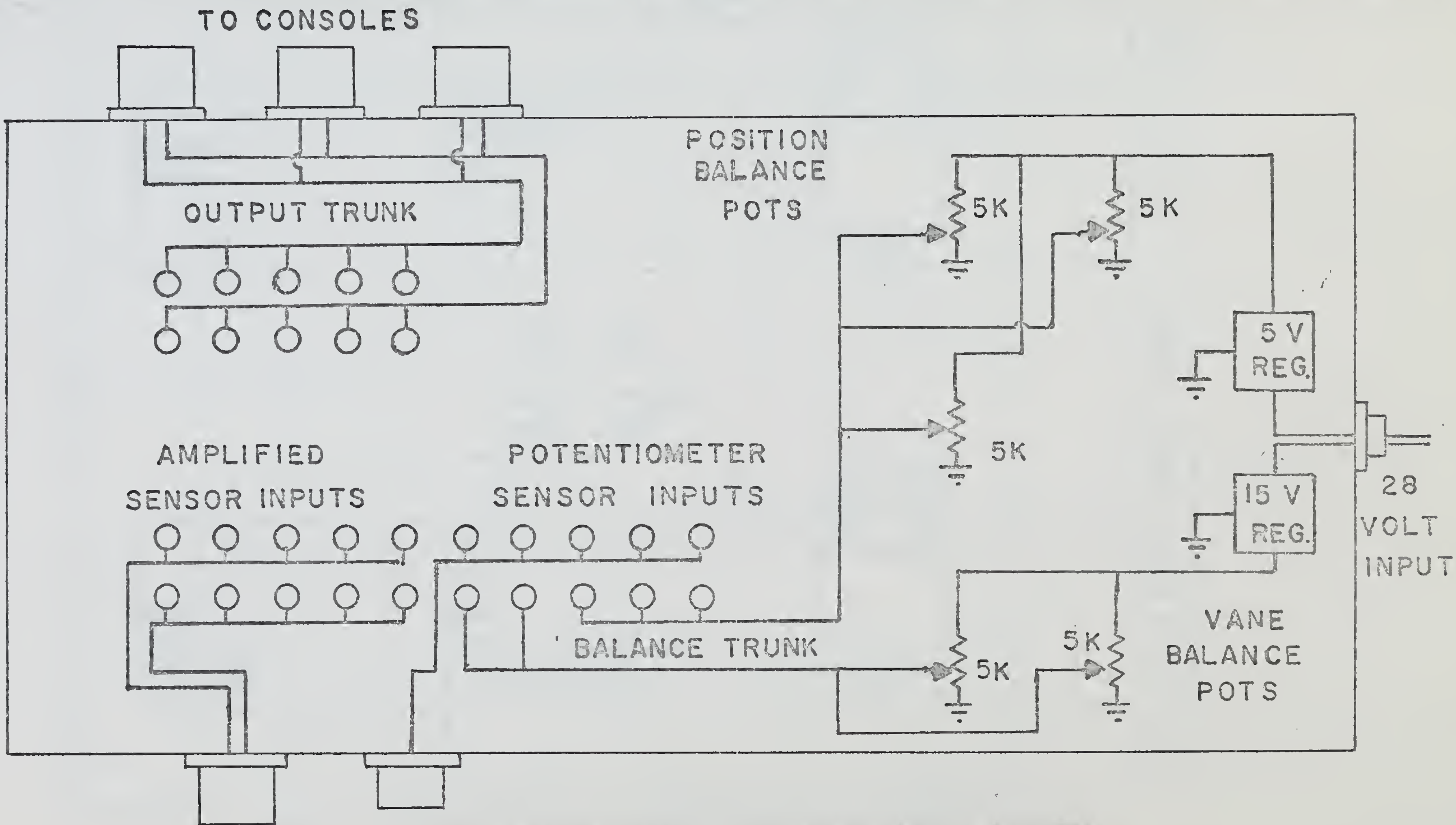


FIGURE 10 J-BOX WIRING DIAGRAM

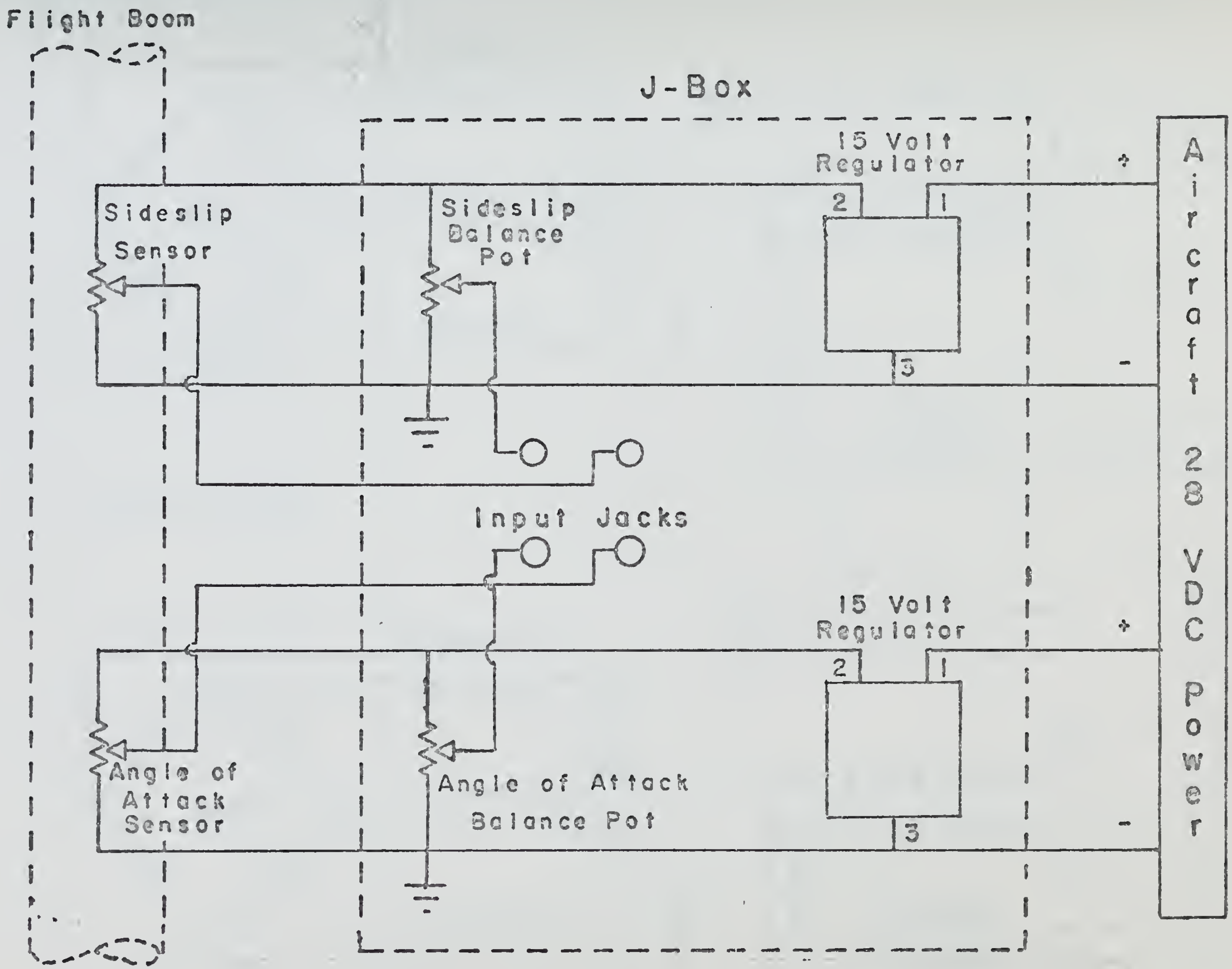


Figure 11 SIDESLIP AND ANGLE OF ATTACK SENSOR CIRCUIT DIAGRAM

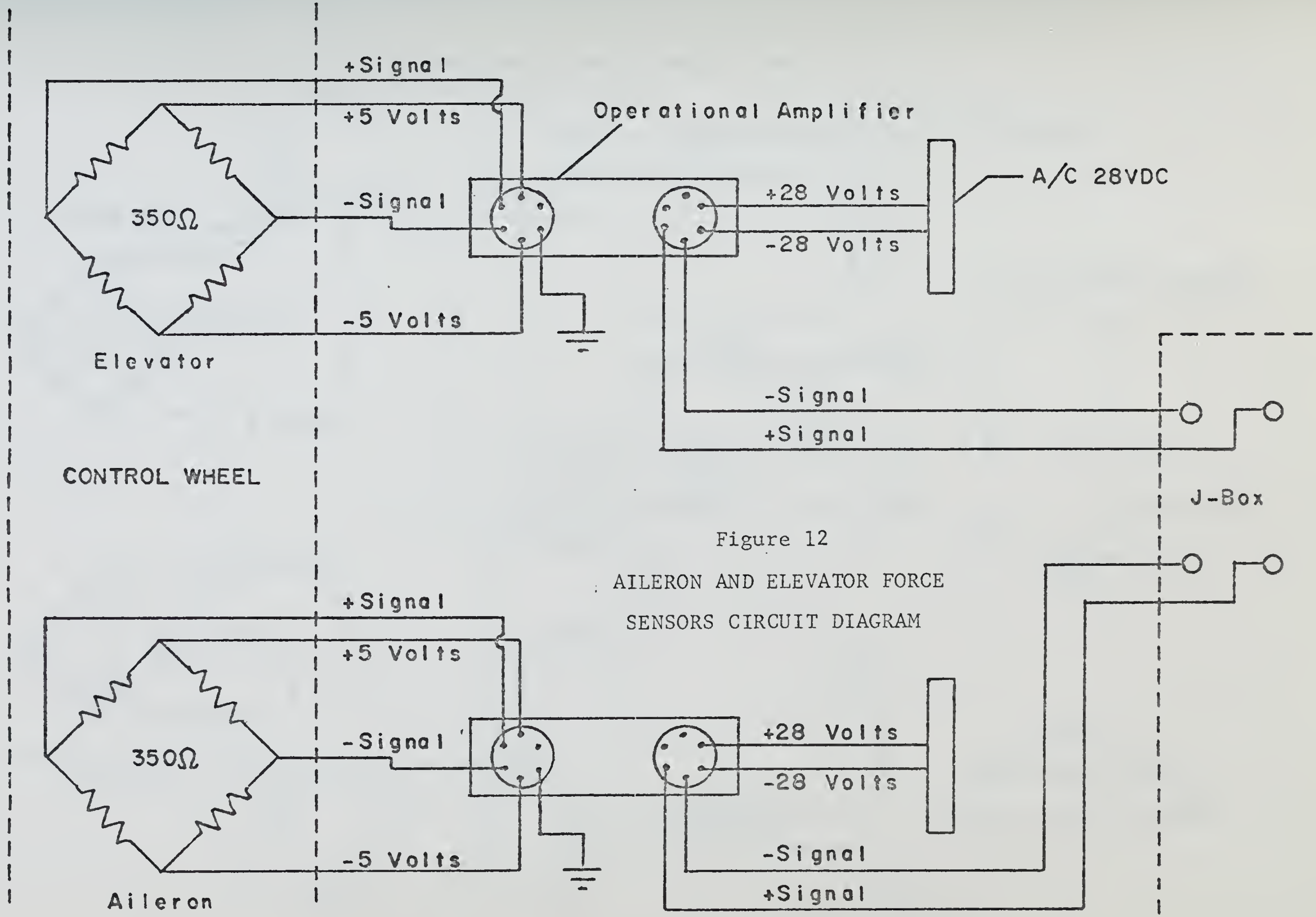


Figure 12

AILERON AND ELEVATOR FORCE

SENSORS CIRCUIT DIAGRAM

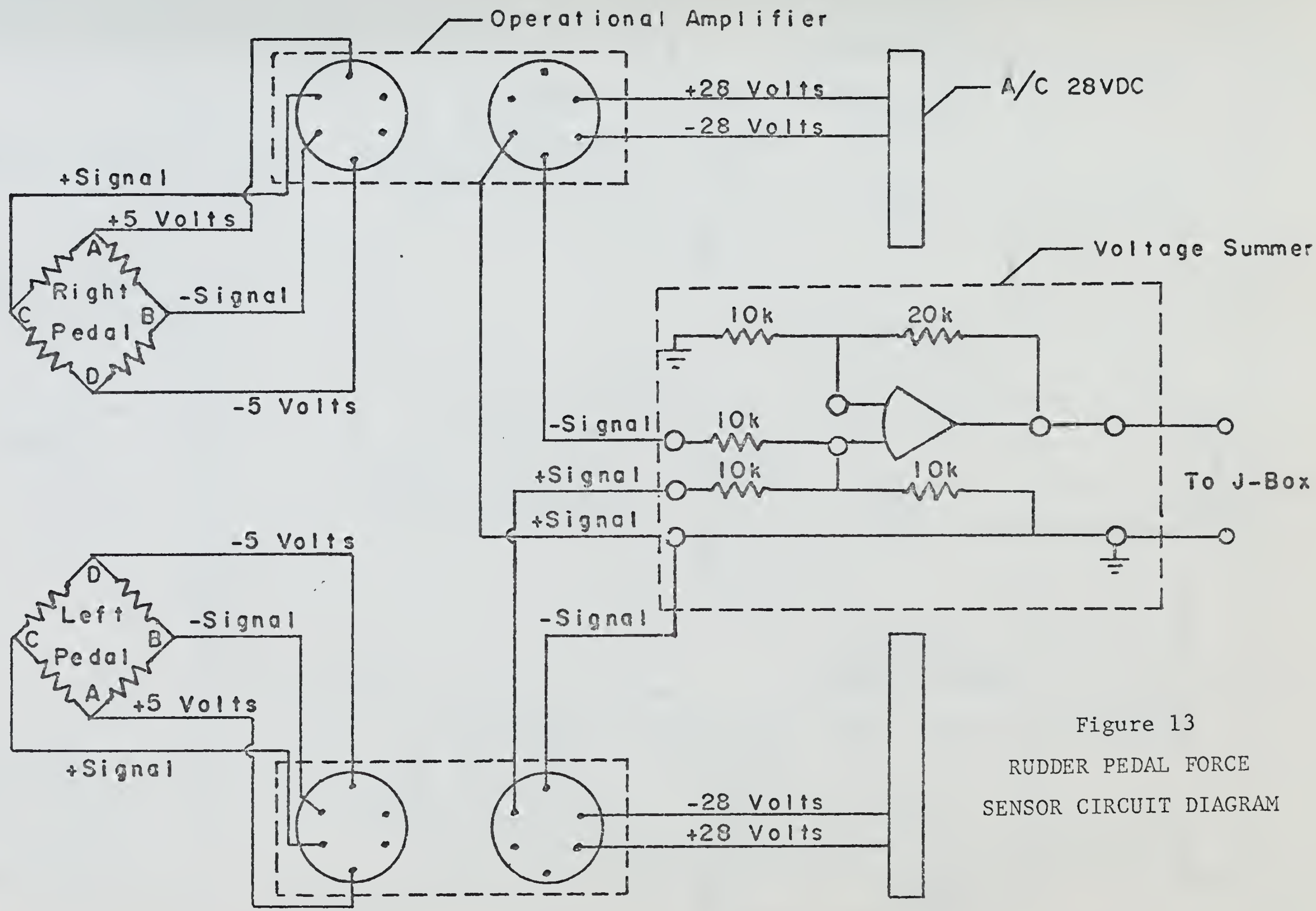


Figure 13

RUDDER PEDAL FORCE

SENSOR CIRCUIT DIAGRAM

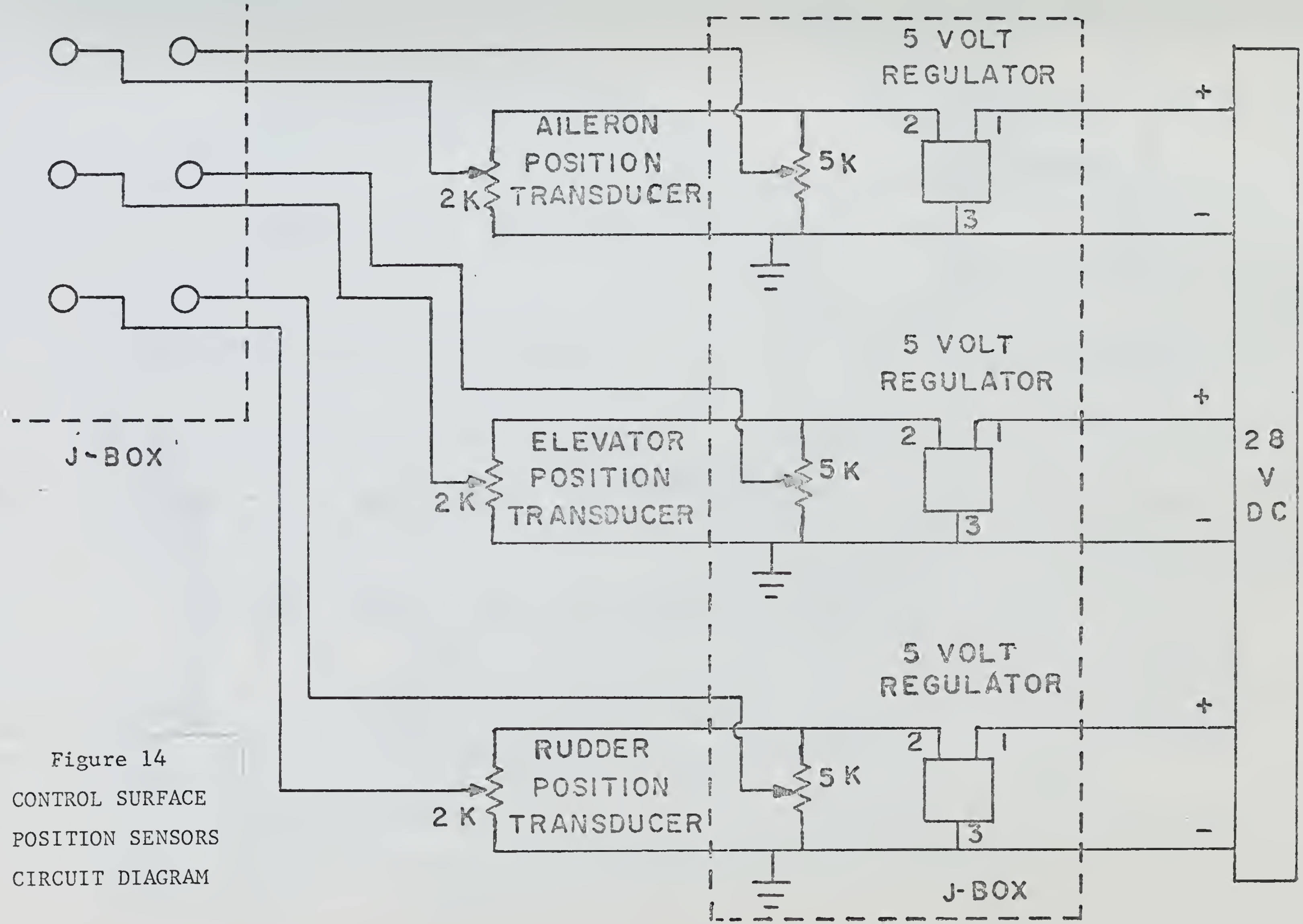


Figure 14
CONTROL SURFACE
POSITION SENSORS
CIRCUIT DIAGRAM

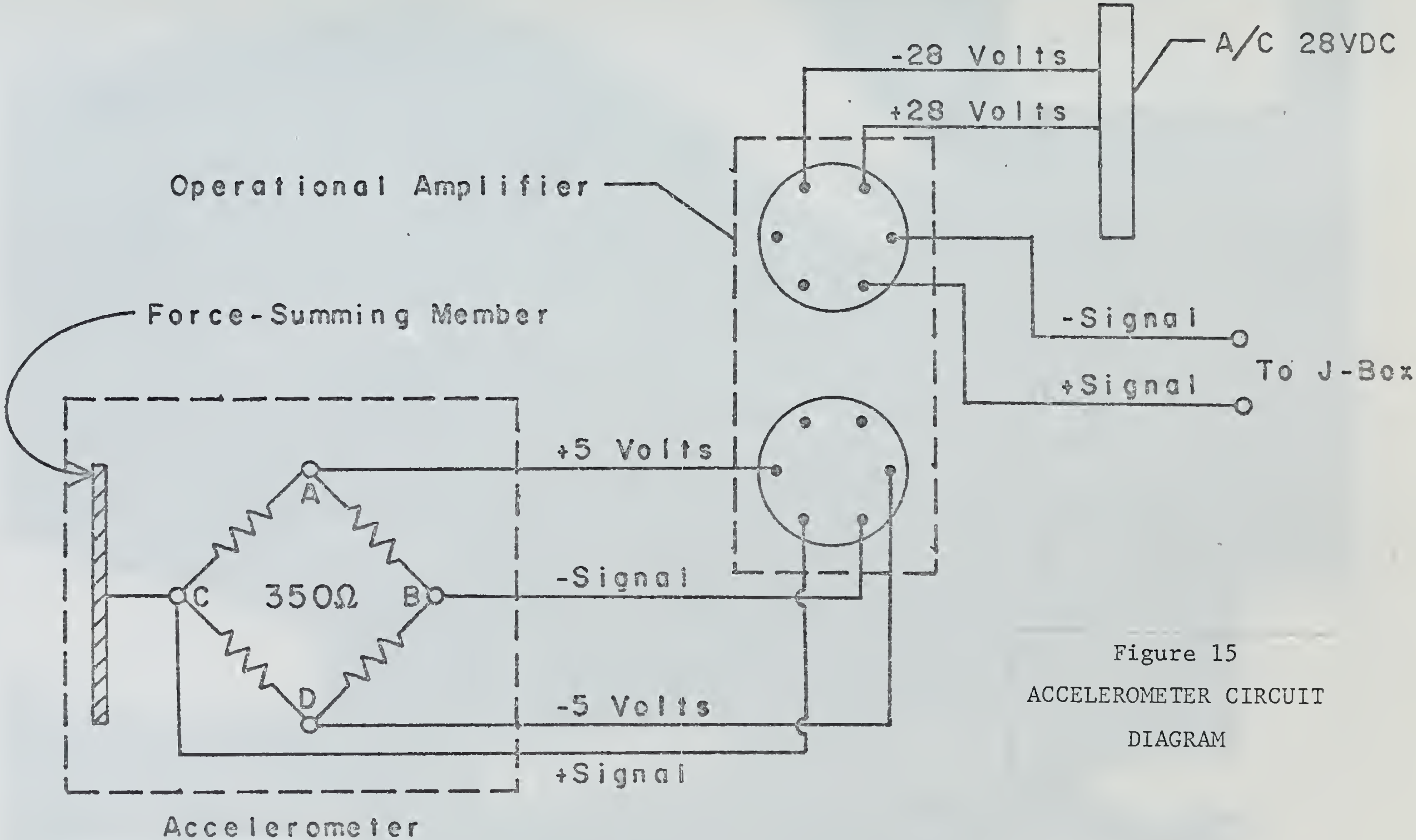


Figure 15
ACCELEROMETER CIRCUIT
DIAGRAM

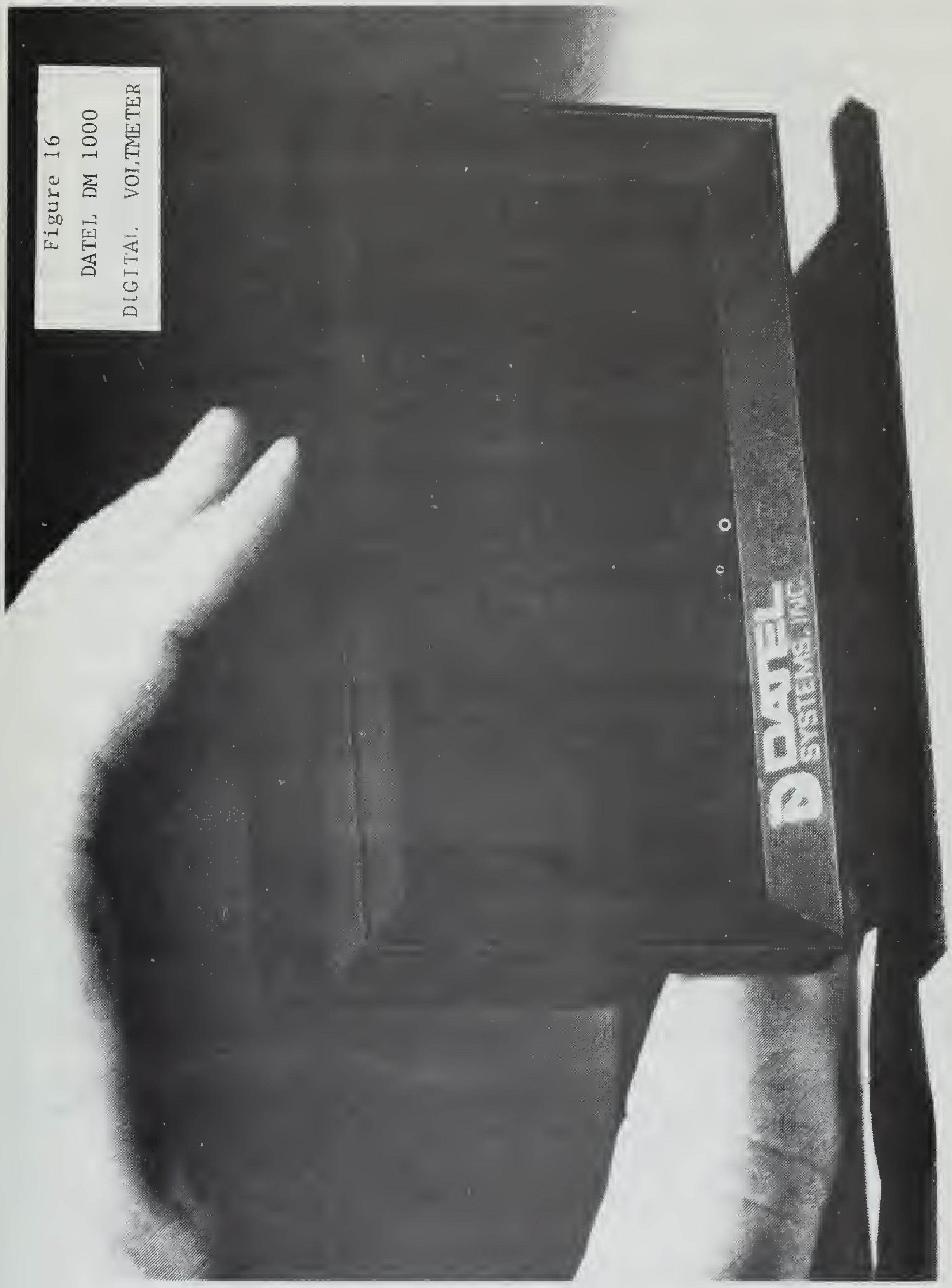


Figure 16
DATEL DM 1000
DIGITAL VOLTMETER

DATEL SYSTEMS, INC.

PARAMETER	MODEL DM-100	MODEL DM-1000
Input		
Input Voltage Range	±199mV ±1.99V	±199.9mV ±1.999V
Input Impedance	>100 MEG OHMS (1)	
Input Bias Current	20 nA (2)	
Input Configuration	Differential	
Input Polarity	Bipolar – Automatic	
Common Mode Rejection	70dB @ 60Hz	
Common Mode Voltage	300V max. to digital output common (3)	
Normal Mode Rejection	40dB @ 60Hz	
Performance		
Accuracy @ 25 C	±0.05% of Reading ±1 Count	
Resolution	10mV (1mV Optional)	1mV (0.1mV Optional)
Temperature Coefficient	50ppm/ C	
Conversion Speed	0 to 1000 Conversions/Second	0 to 200 Conversions/Second
Input Settling Time	50 μsec for a F S Change W O Input Filter (6)	
Operating Temperature Range	0 C to +60 C (4)	
Storage Temperature Range	–55 C to +85 C	
Warm Up Time	5 Minutes to Specified Accuracy	
Adjustments	Zero, Balance, Full Scale Located Behind Snap On Front Bezel	
Input Power	5VDC ±0.25VDC @ 750mA (115VAC ±10VAC, 50-450Hz, Optional)	
Display Output		
Display Type	Solid State LED for Data Digits, 100% Overrange, Overflow, Decimal point and Polarity – Character Height .27 in	
Overflow	Indicated by the Letters "OF"	
Decimal Points	Selectable at rear Connector	
Data Outputs (5)		
BCD Outputs	8 Parallel Lines, BCD (8–4–2–1) Positive Logic Loading: 2TTL Loads	12 Parallel Lines, BCD (8–4–2–1) Positive Logic Loading: 2TTL Loads
Overrange	>100 counts (DM-100) or >1000 counts (DM-1000) indicated with a HIGH Loading: 2 TTL loads*	
Polarity	Input signal polarity indicated with a HIGH-positive, LOW-negative. Loading: 2 TTL loads*	
Overflow (OF)	HIGH-input signal within range LOW-input signal outside range. Loading: 2 TTL loads*	
End of Conversion (EOC)	HIGH – During the reset and conversion period LOW – Conversion complete. Loading: 2 TTL loads*	
Input/Output Control (5)		
External Start Conversion Command	Positive pulse 100 nsec. min. Transition from "LOW" to "HIGH" resets output register and blanks readout. The conversion process is initiated upon return from "HIGH" to "LOW". Loading: 1 TTL load*	
Internal Start Gate	Controls internal start clock "HIGH" – Run "LOW" – Stop loading: 1 TTL load*	
Internal Start Adjust	Controls Rate of Internal Start Clock – see Applications Section	
Internal Start Out	Positive Pulse Output of Internal Start Clock – see Applications Section	
Lamp Test Input	Grounding this input figure +1888 or +188 is displayed for testing all display segments. Loading: Sink 35mA	
Decimal Point Inputs (DP1, DP10, DP100)	Grounding inputs illuminates corresponding decimal points on the display Loading: Sink 15mA	
Physical		
Case Size	3"W X 1.75"H X 2.25"D	
Case Material	Black LEXAN	
Weight	6 oz. Approx.	
Mounting	Through a 1.75" X 3.00" Cut-Out and Secured with Four 4-40 Tapped Holes	
I/O Connector	36 Contact, Edgeboard Type Viking – 3VH18/1JHD-5 – Wire Wrap 3VH18/1JN-5 – Solder Tab	

Notes:
(1) 1000 MEG OHM Standard When 1nA Input Bias Current is Ordered
(2) 1nA (Optional)
(3) 300 Volts Common Mode is Standard When Optical Isolator is Ordered otherwise the Max. CMV is ±2V.
(4) –25°C to +85°C (Optional)
(5) DTL/TTL Compatible
(6) DM-1000 – 1.4 sec. / (6) DM-100 – 1.0 sec. } with normal mode filter
*Low: V out ("0") = ≤+0.8V. HIGH: V out ("1") = ≥+2.4V.

Figure 17

DATEL DM 1000 DIGITAL VOLTMETER SPECIFICATIONS

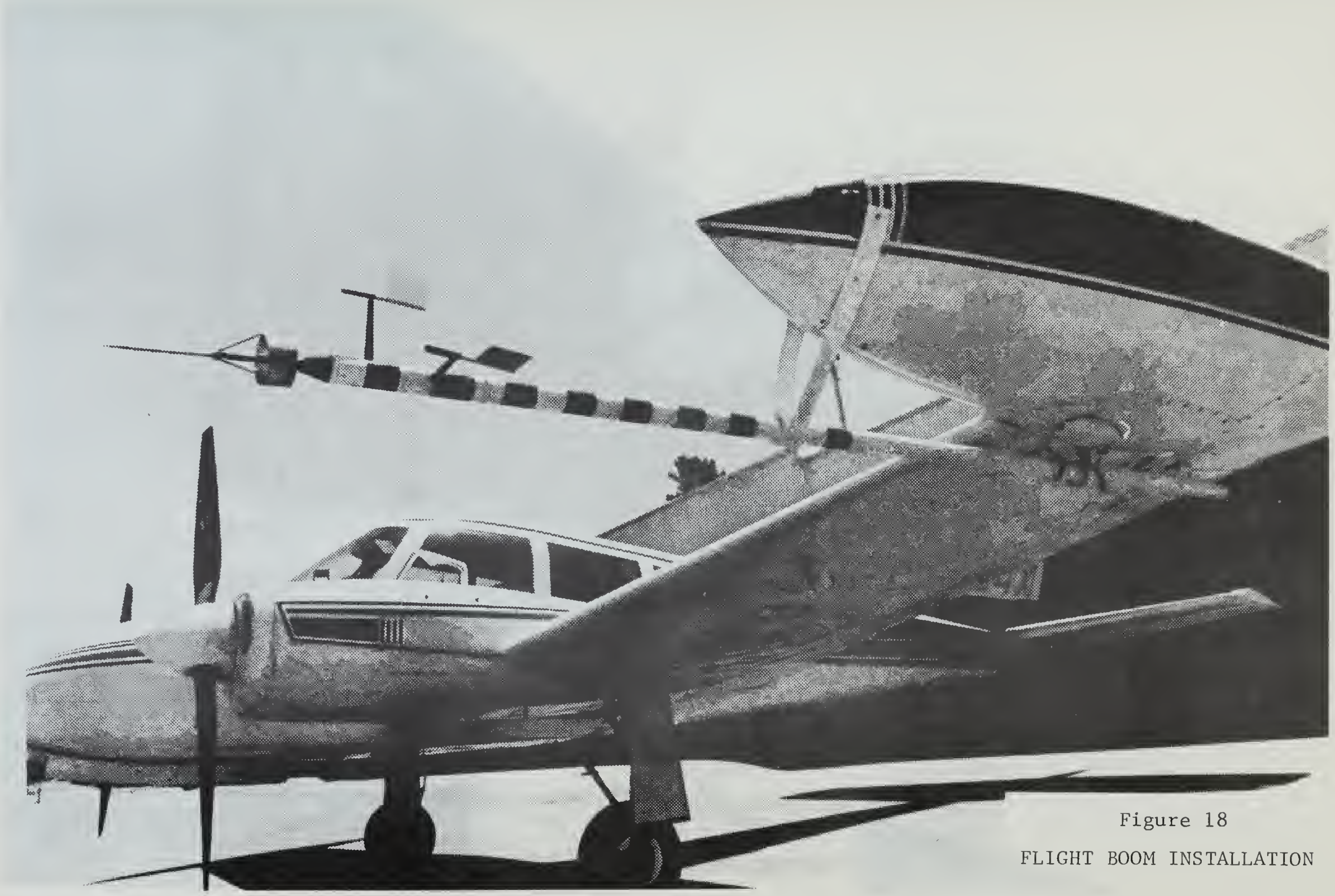


Figure 18

FLIGHT BOOM INSTALLATION

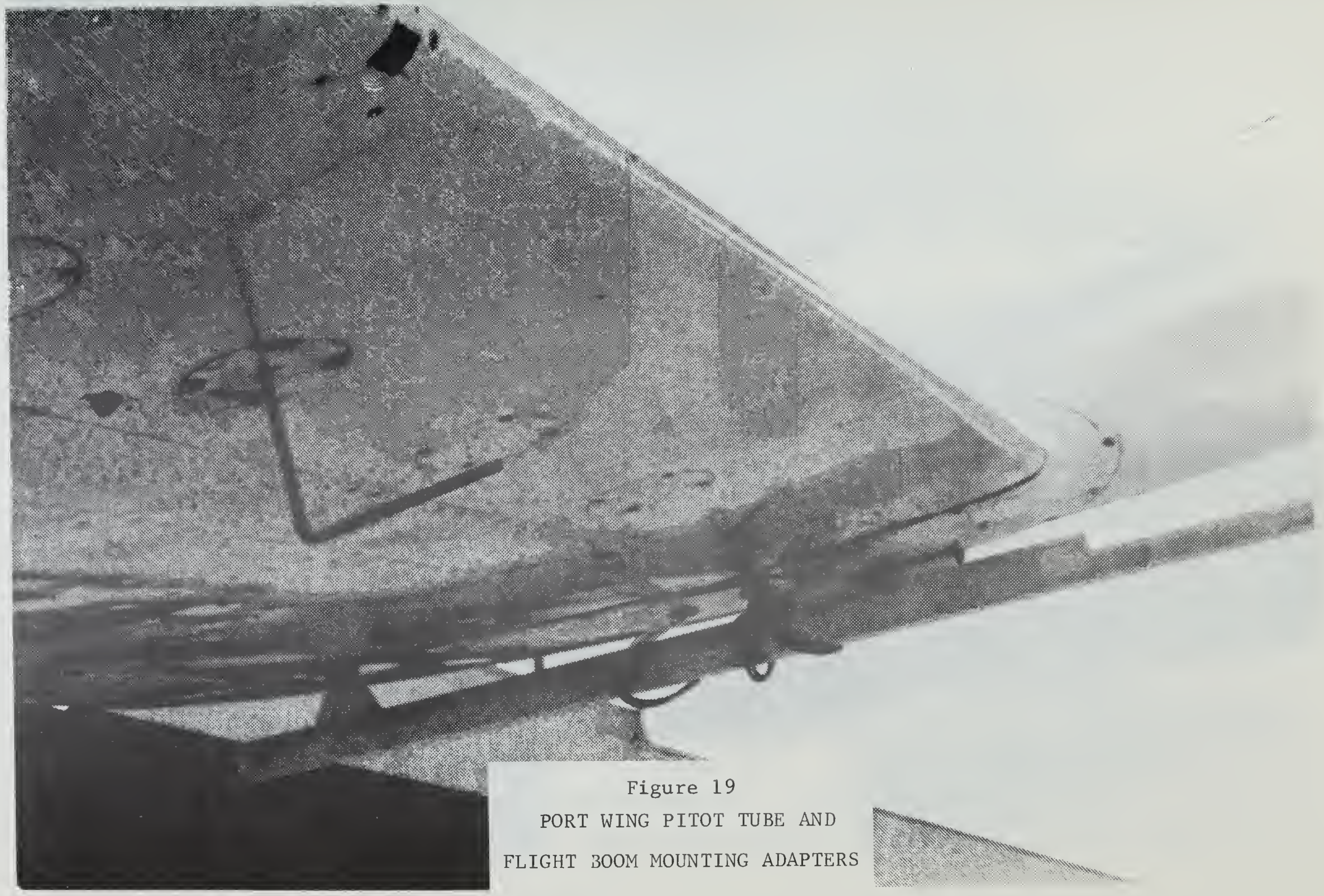


Figure 19
PORT WING PITOT TUBE AND
FLIGHT BOOM MOUNTING ADAPTERS

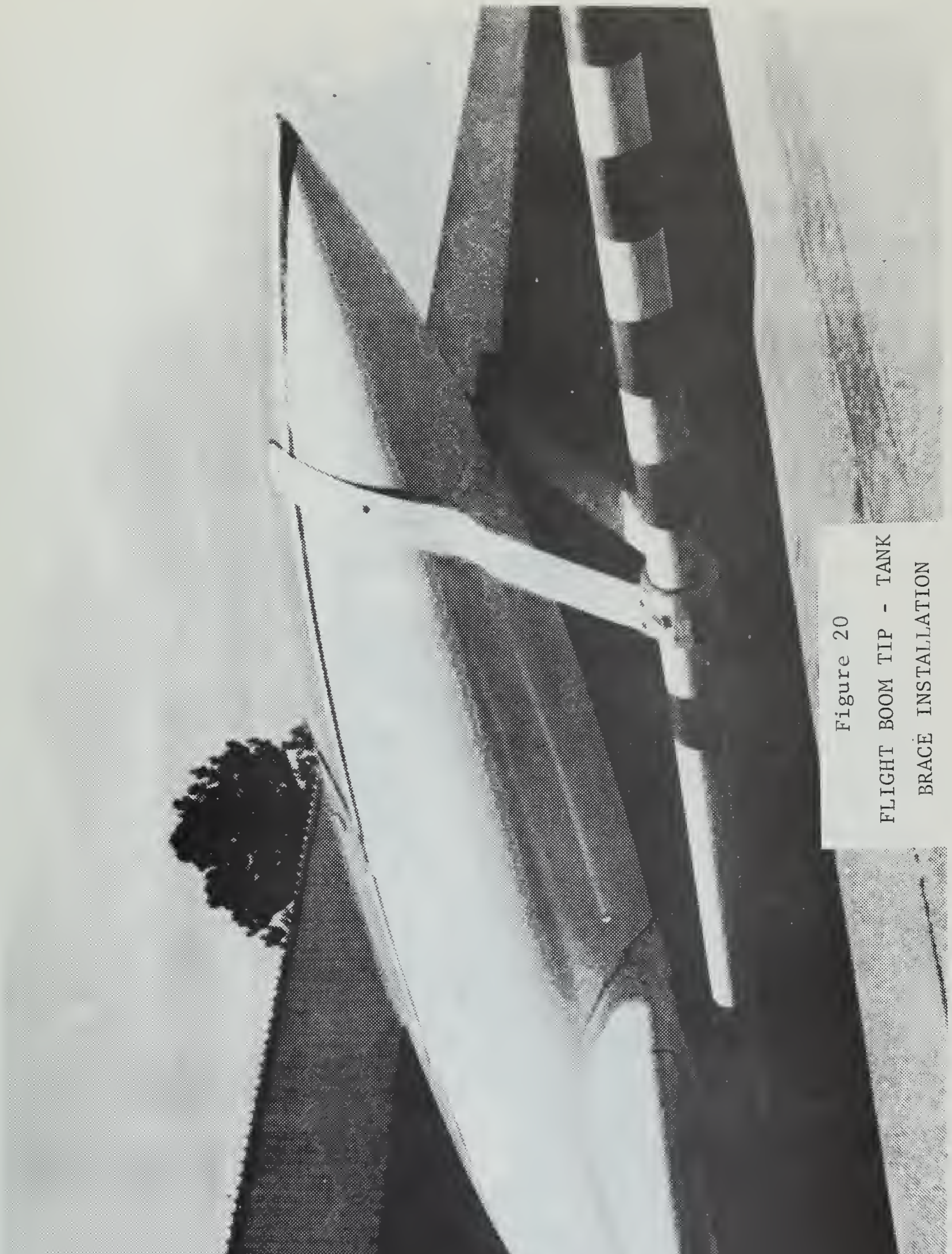


Figure 20
FLIGHT BOOM TIP - TANK
BRACE INSTALLATION

Figure 21
FLIGHT BOOM WING SPAR
MOUNTING BRACKETS

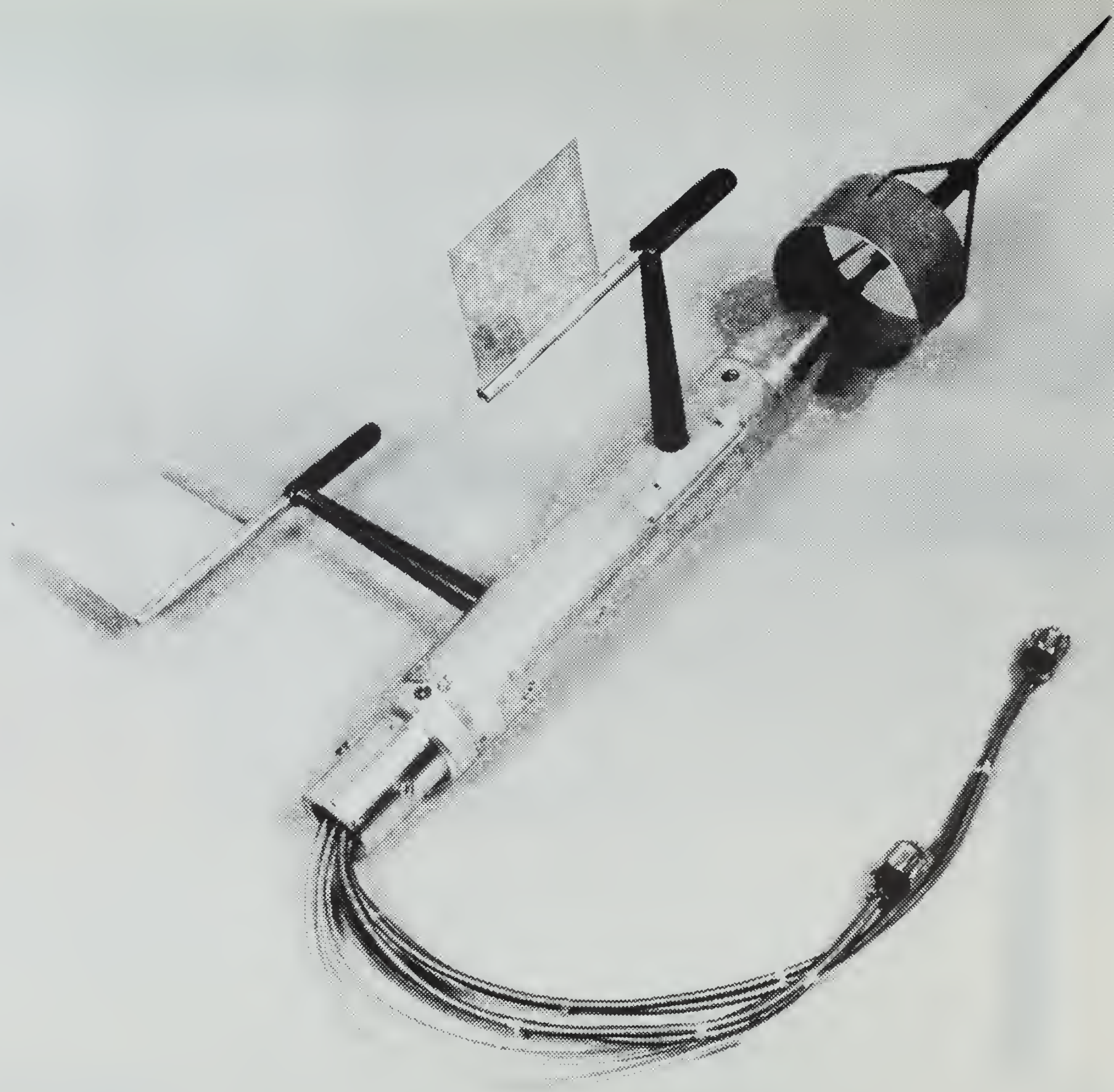


Figure 22
YAW AND PITCH SENSING
(YAPS) HEAD

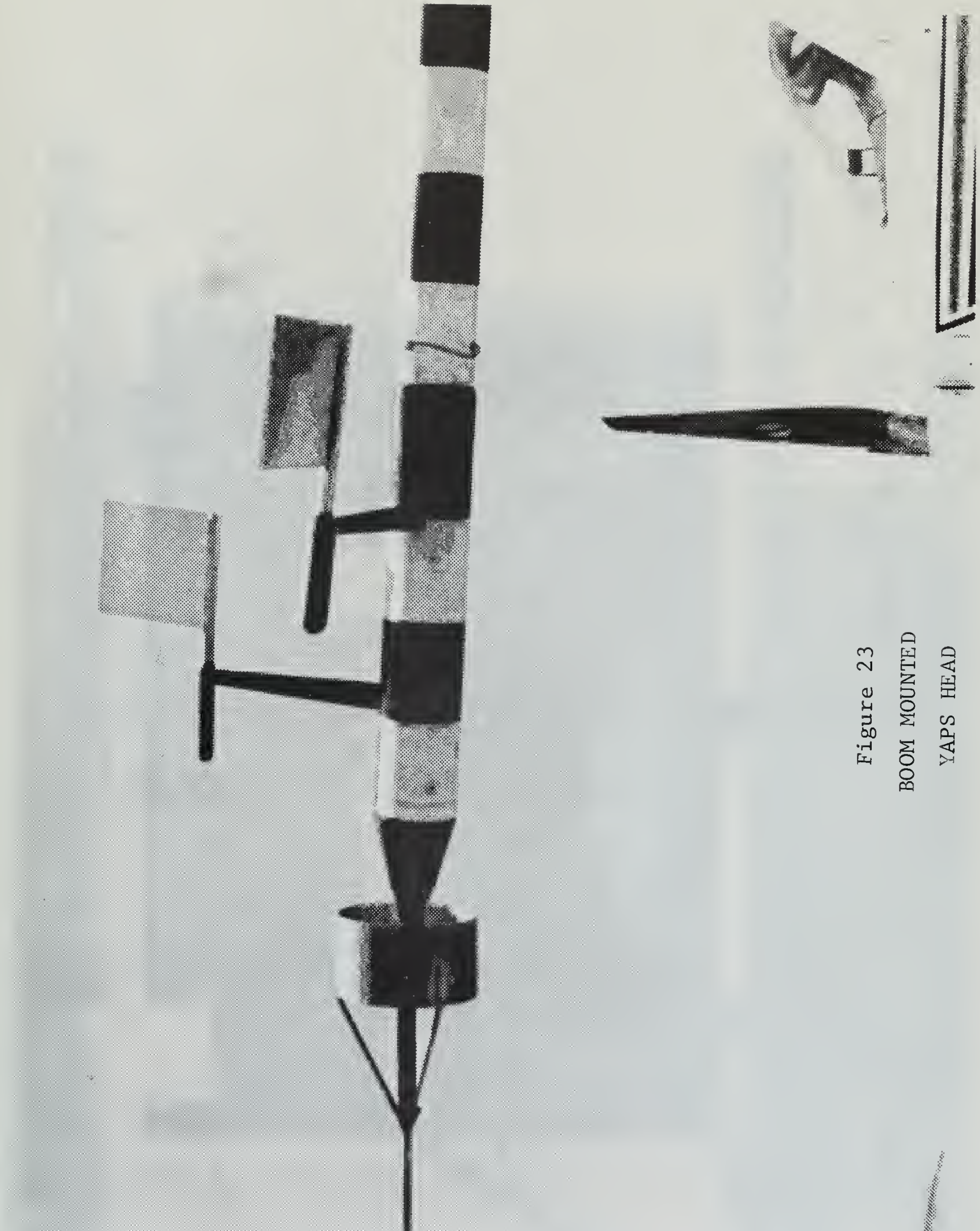


Figure 23
BOOM MOUNTED
YAPS HEAD

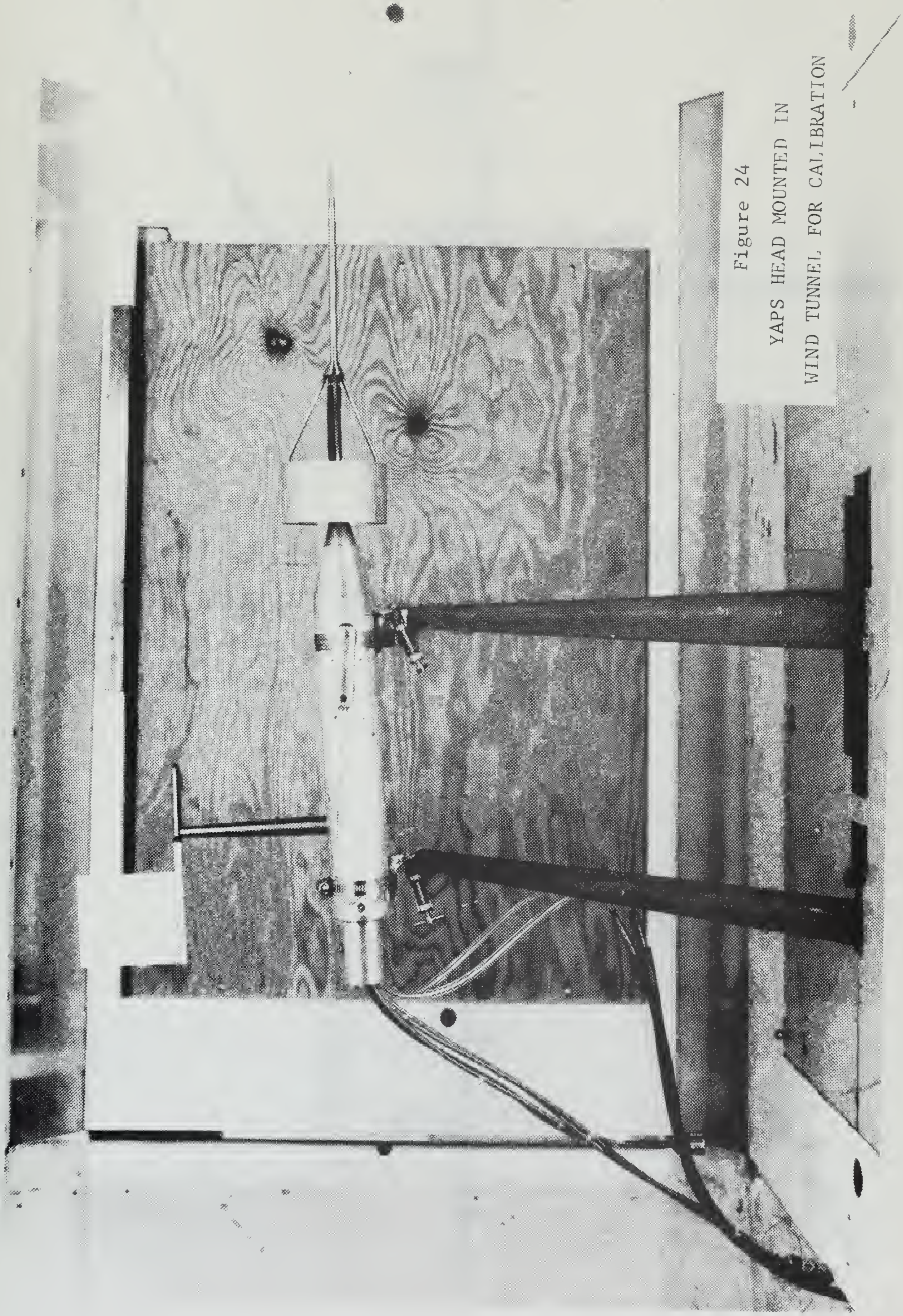


Figure 24

YAPS HEAD MOUNTED IN
WIND TUNNEL FOR CALIBRATION

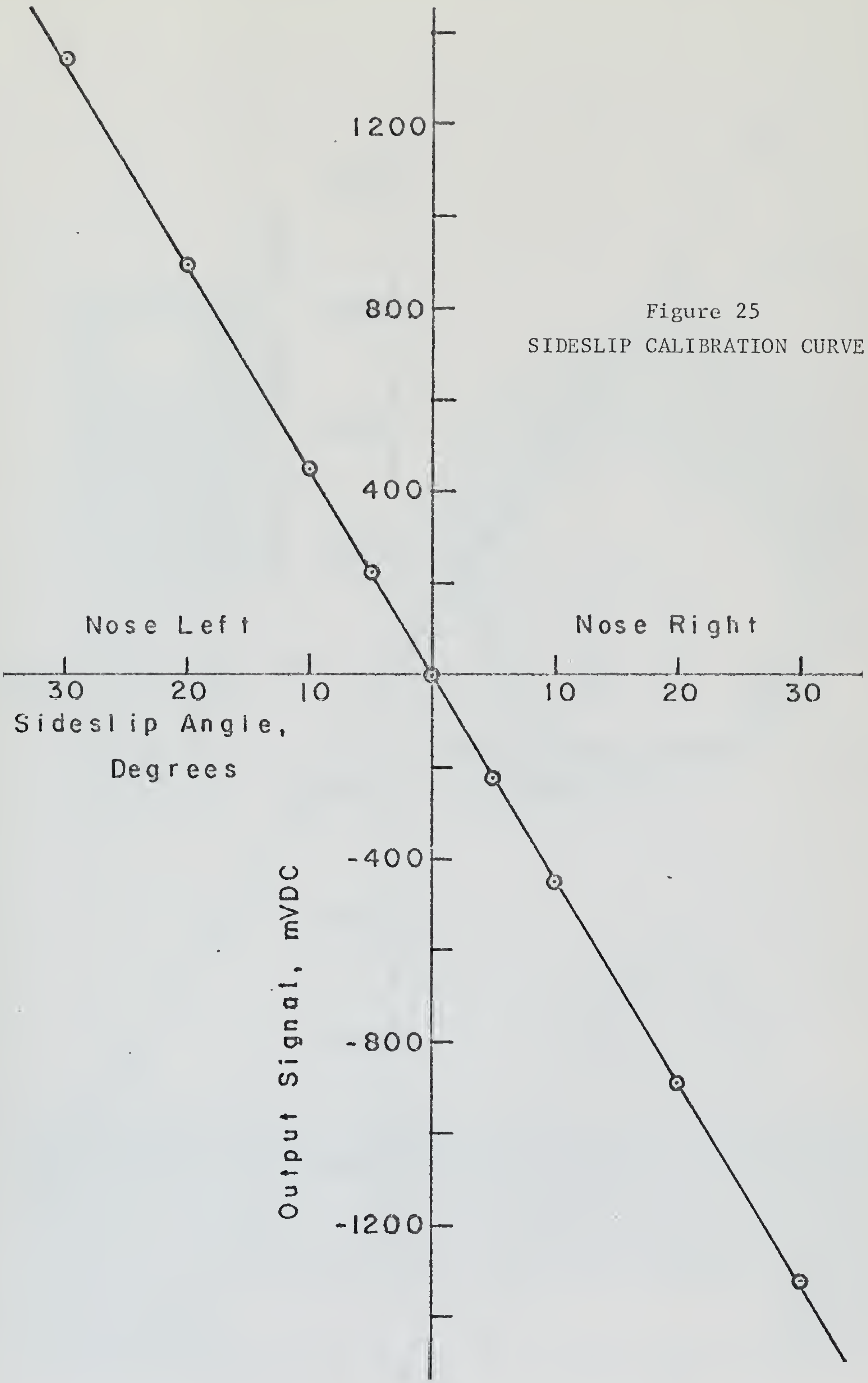


Figure 25
SIDESLIP CALIBRATION CURVE

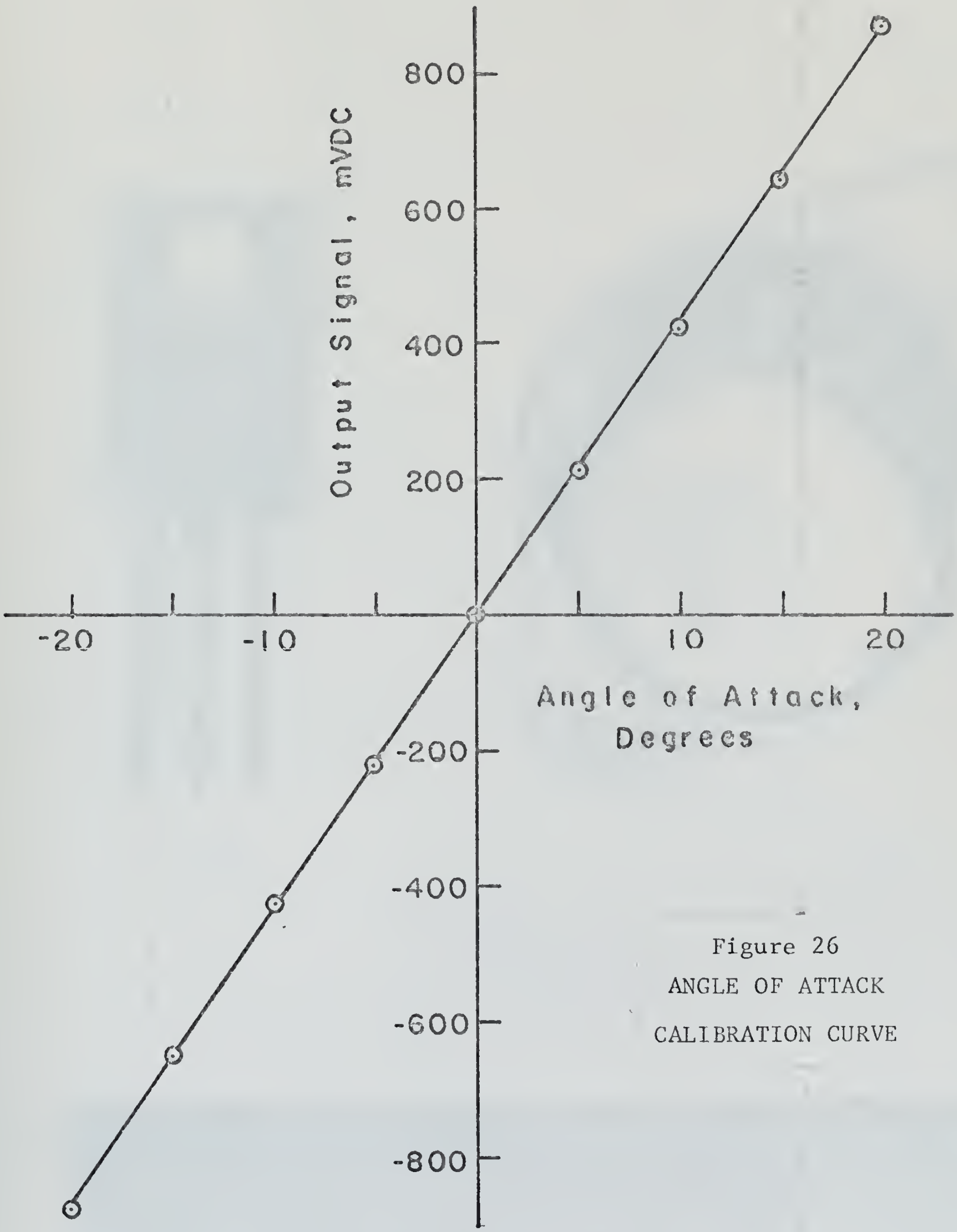
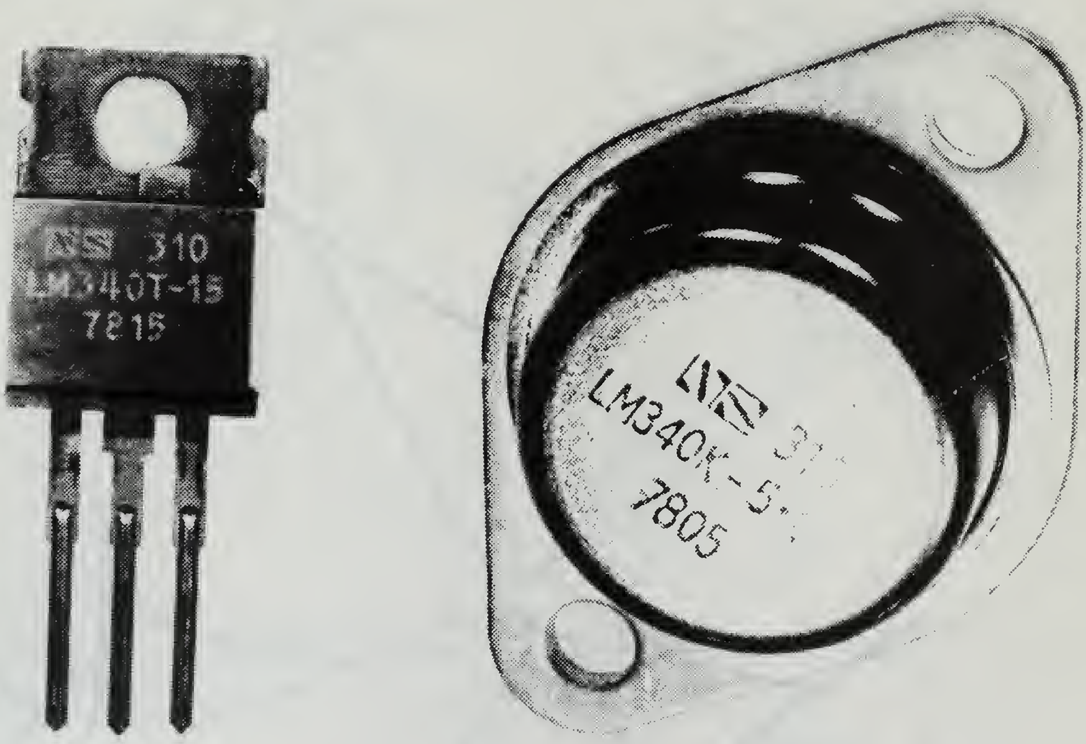


Figure 26
ANGLE OF ATTACK
CALIBRATION CURVE



15 VOLT

5 Volt

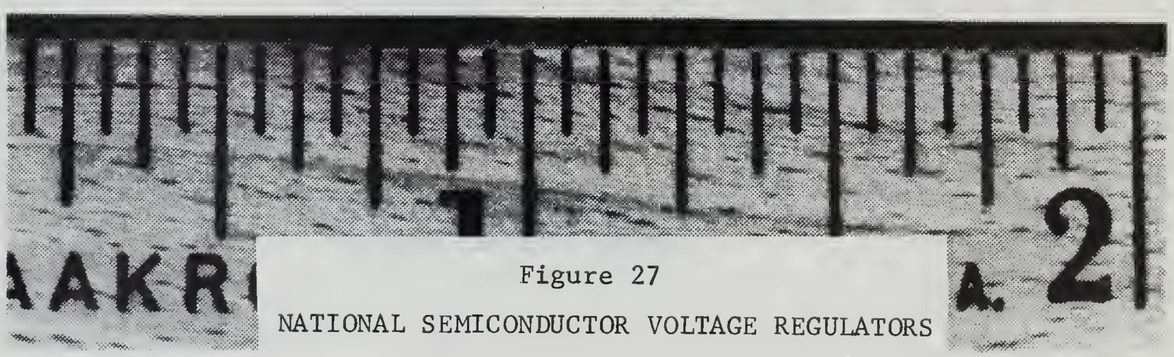


Figure 27
NATIONAL SEMICONDUCTOR VOLTAGE REGULATORS

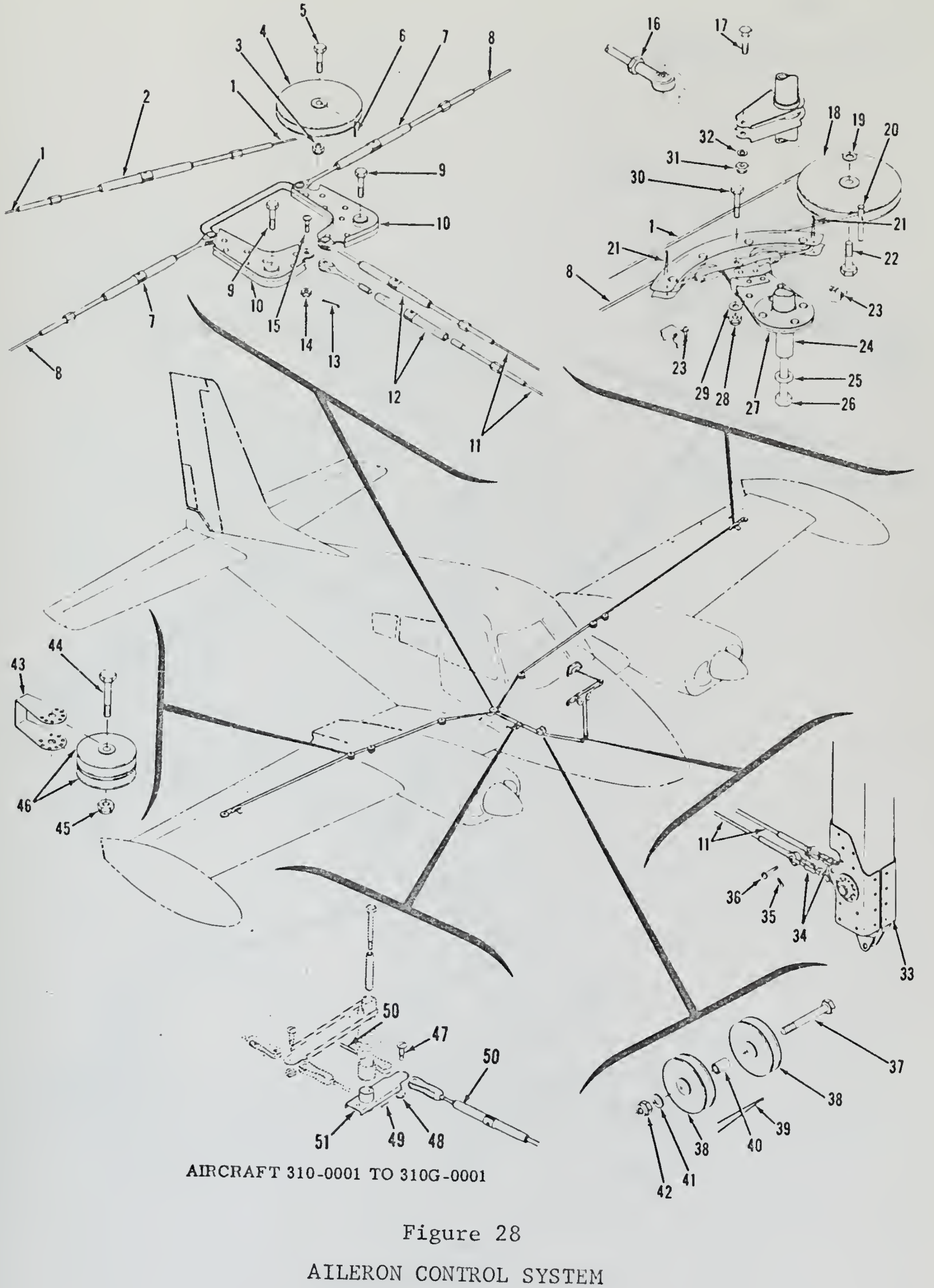


Figure 28

AILERON CONTROL SYSTEM

NOTE

Minimum torque on bolt (56) is 20 pound-inches.

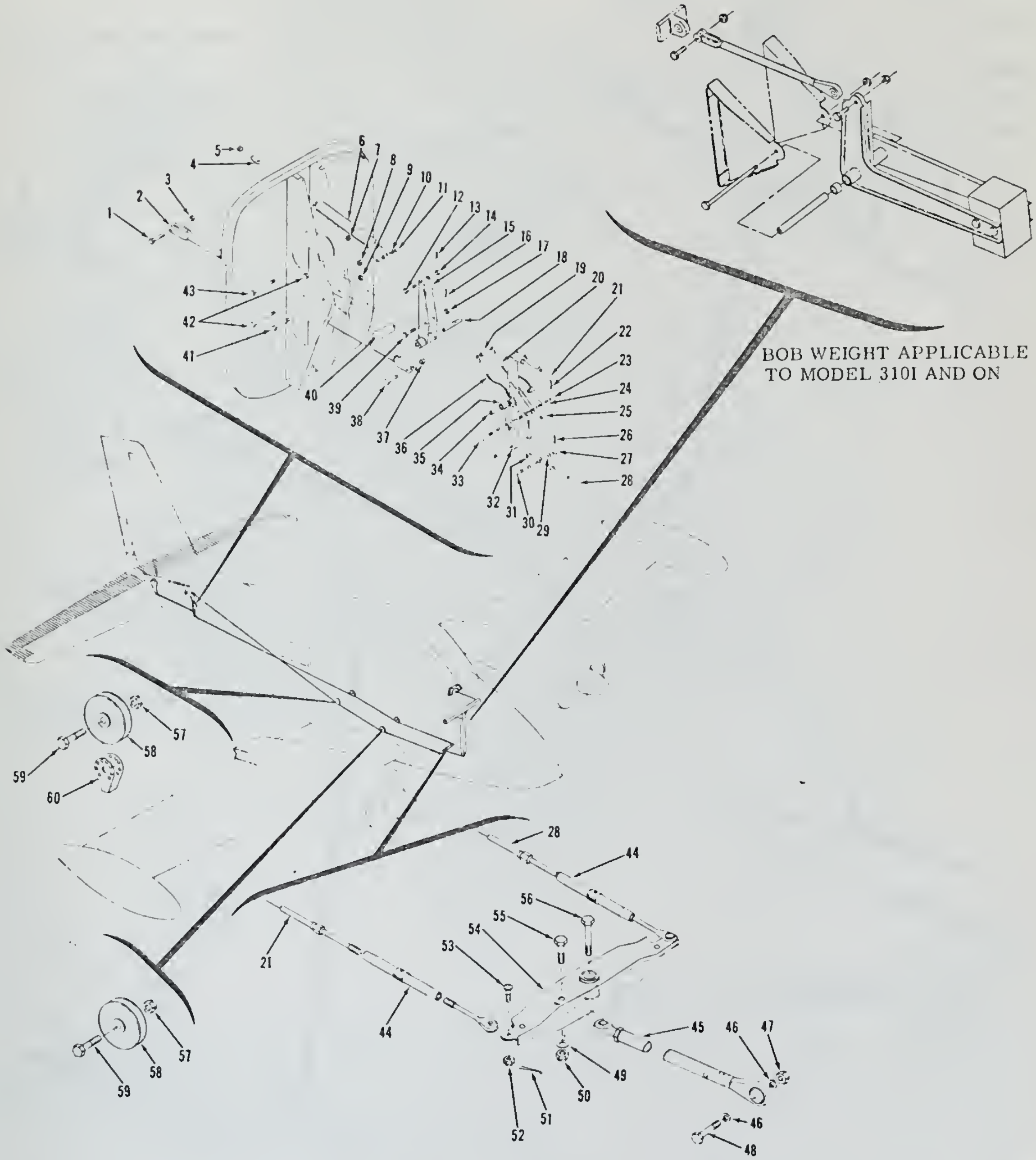


Figure 29

ELEVATOR CONTROL SYSTEM

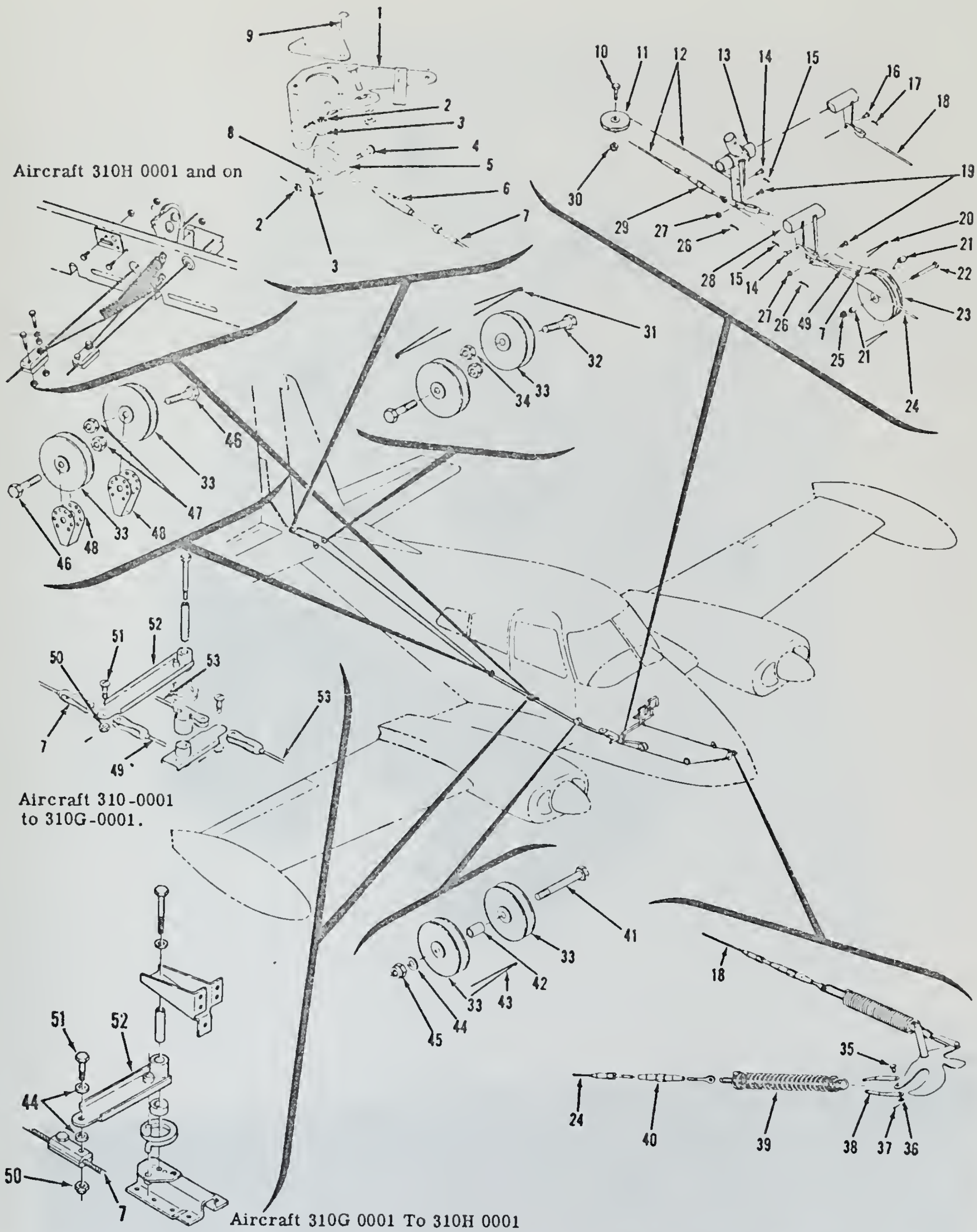


Figure 30

RUDDER CONTROL SYSTEM

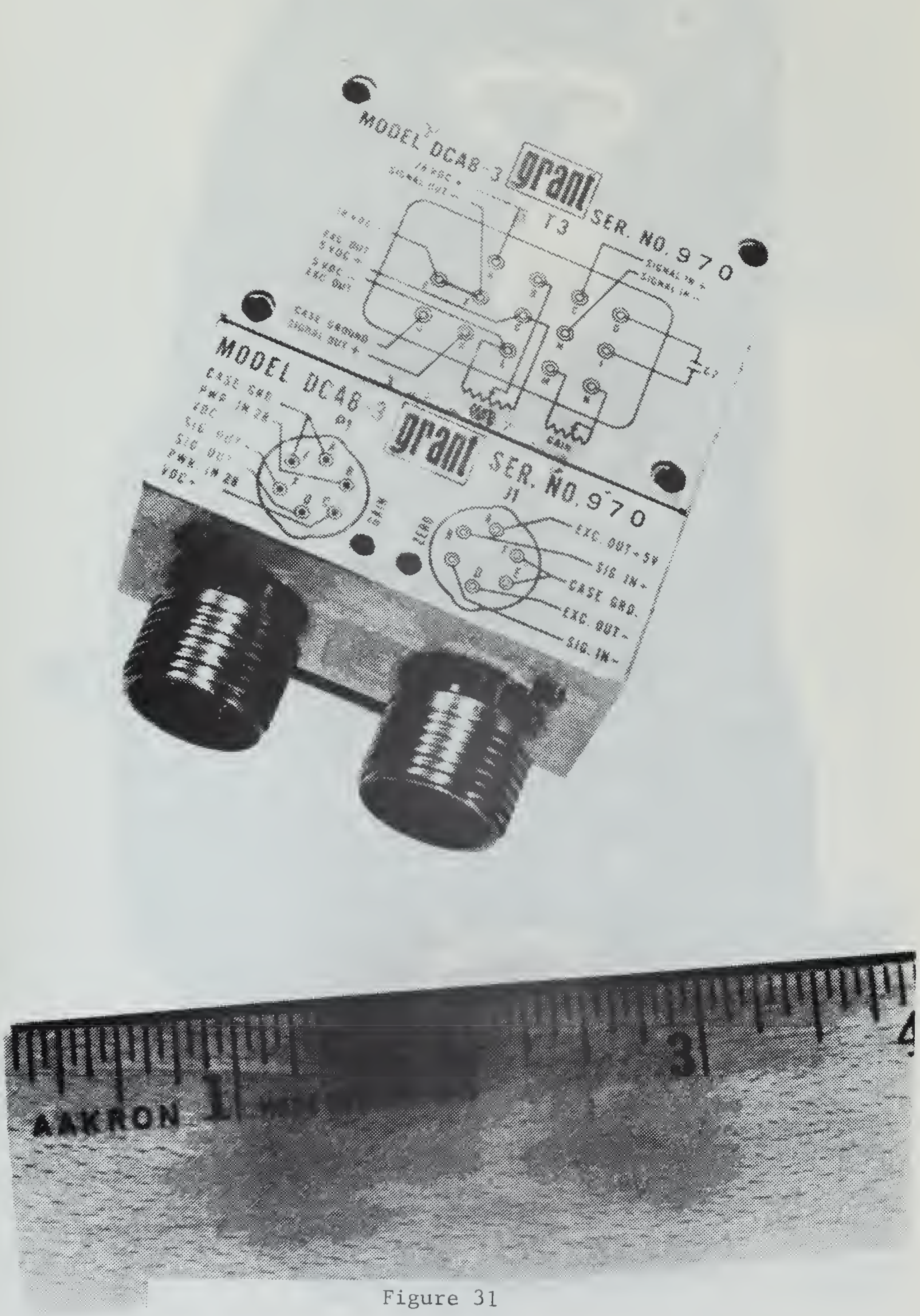


Figure 31

GRANT MODEL DCA8-3 OPERATIONAL AMPLIFIER

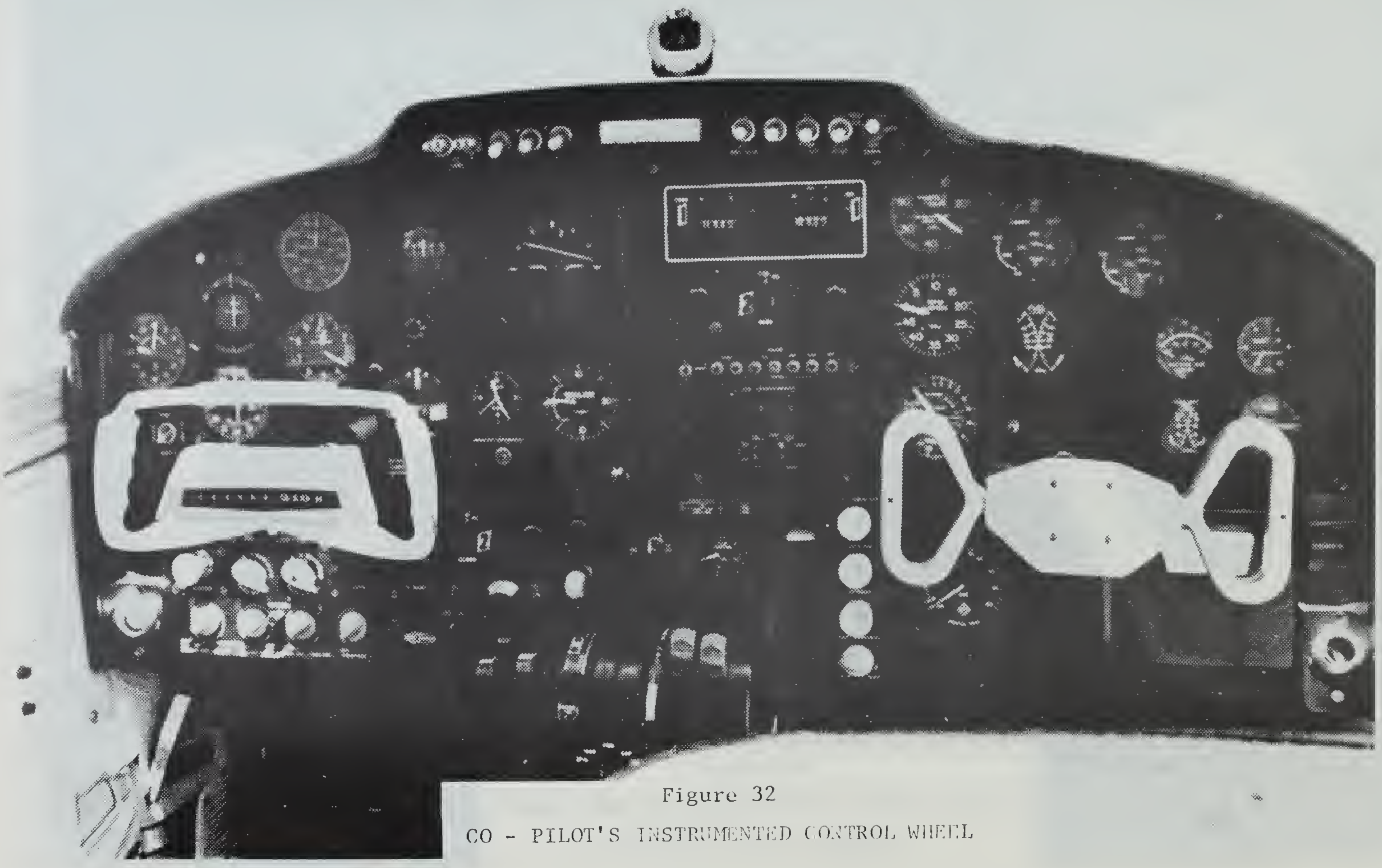


Figure 32

CO - PILOT'S INSTRUMENTED CONTROL WHEEL

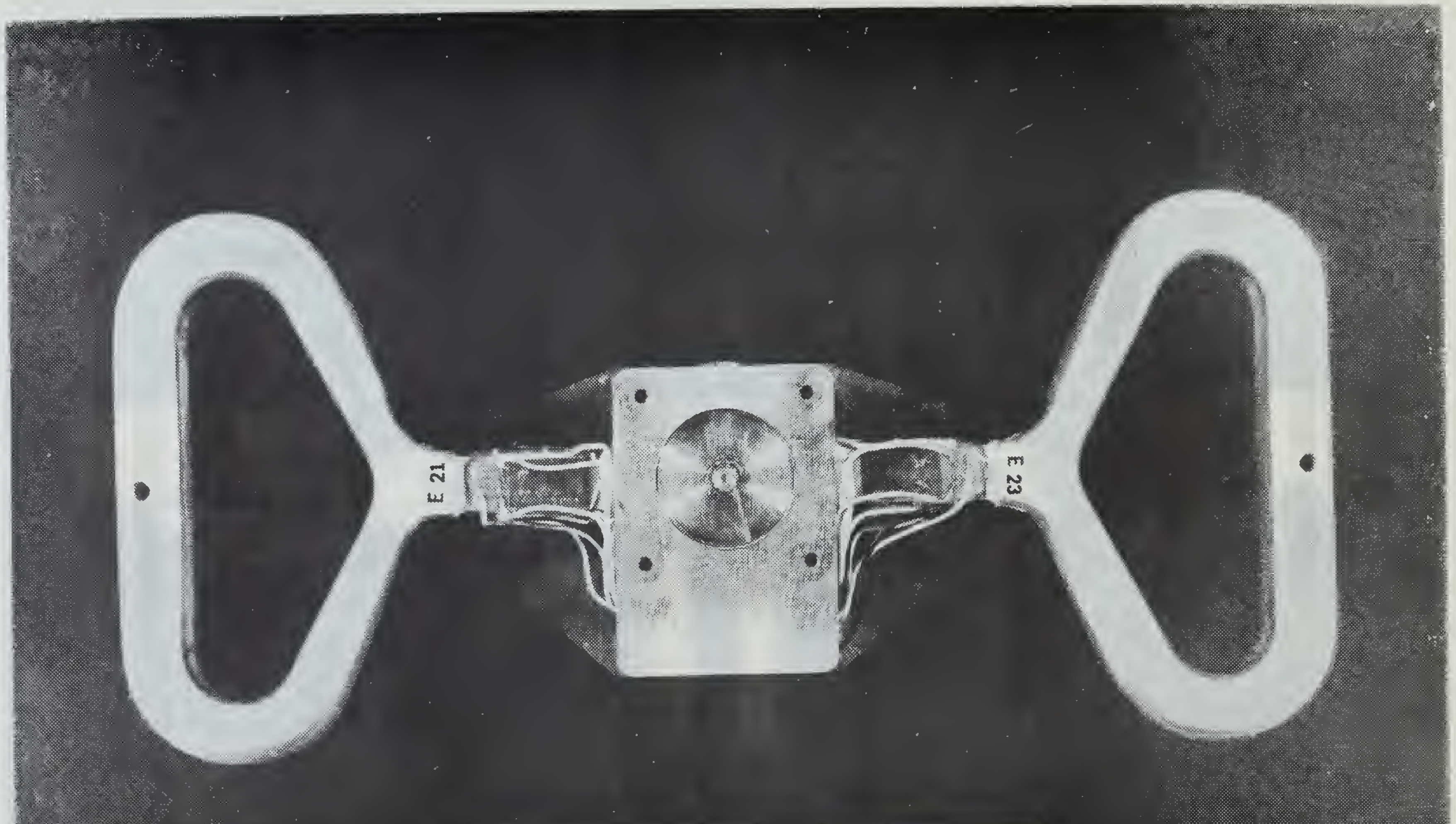


Figure 33

FRONT VIEW OF INSTRUMENTED CONTROL WHEEL

WITH COVER PLATE REMOVED

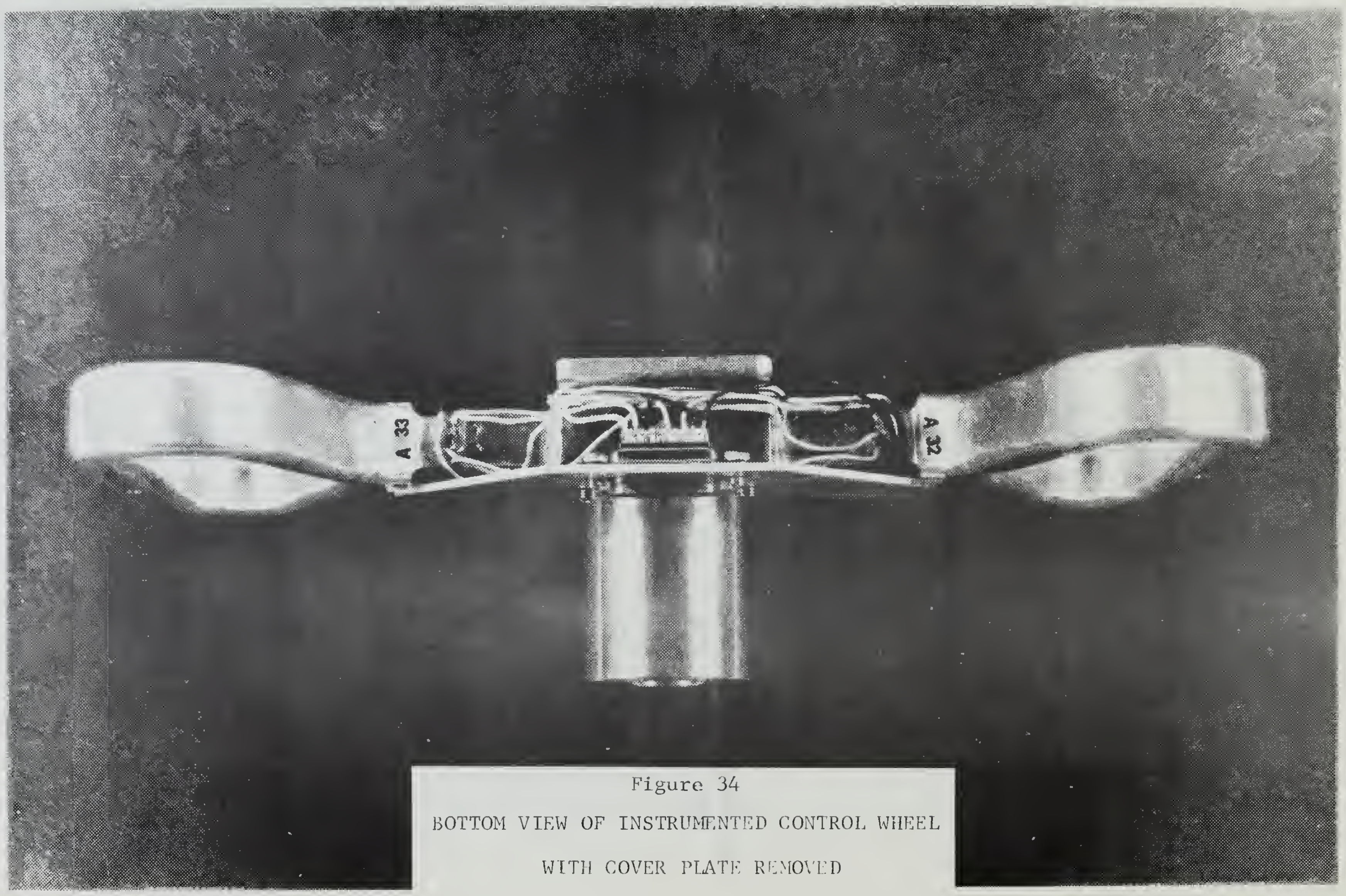


Figure 34

BOTTOM VIEW OF INSTRUMENTED CONTROL WHEEL

WITH COVER PLATE REMOVED

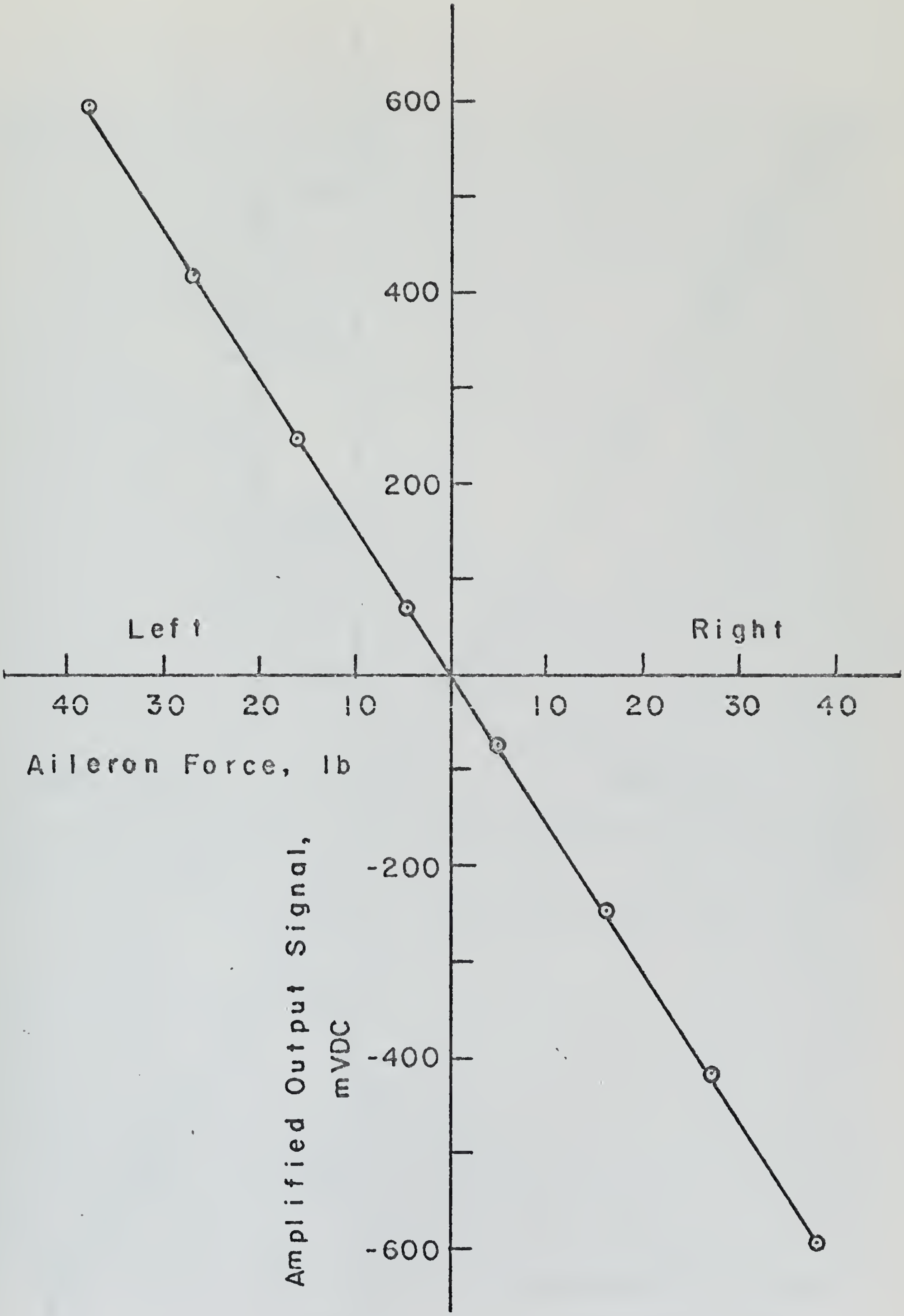


Figure 35

AILERON FORCE SENSOR
CALIBRATION CURVE

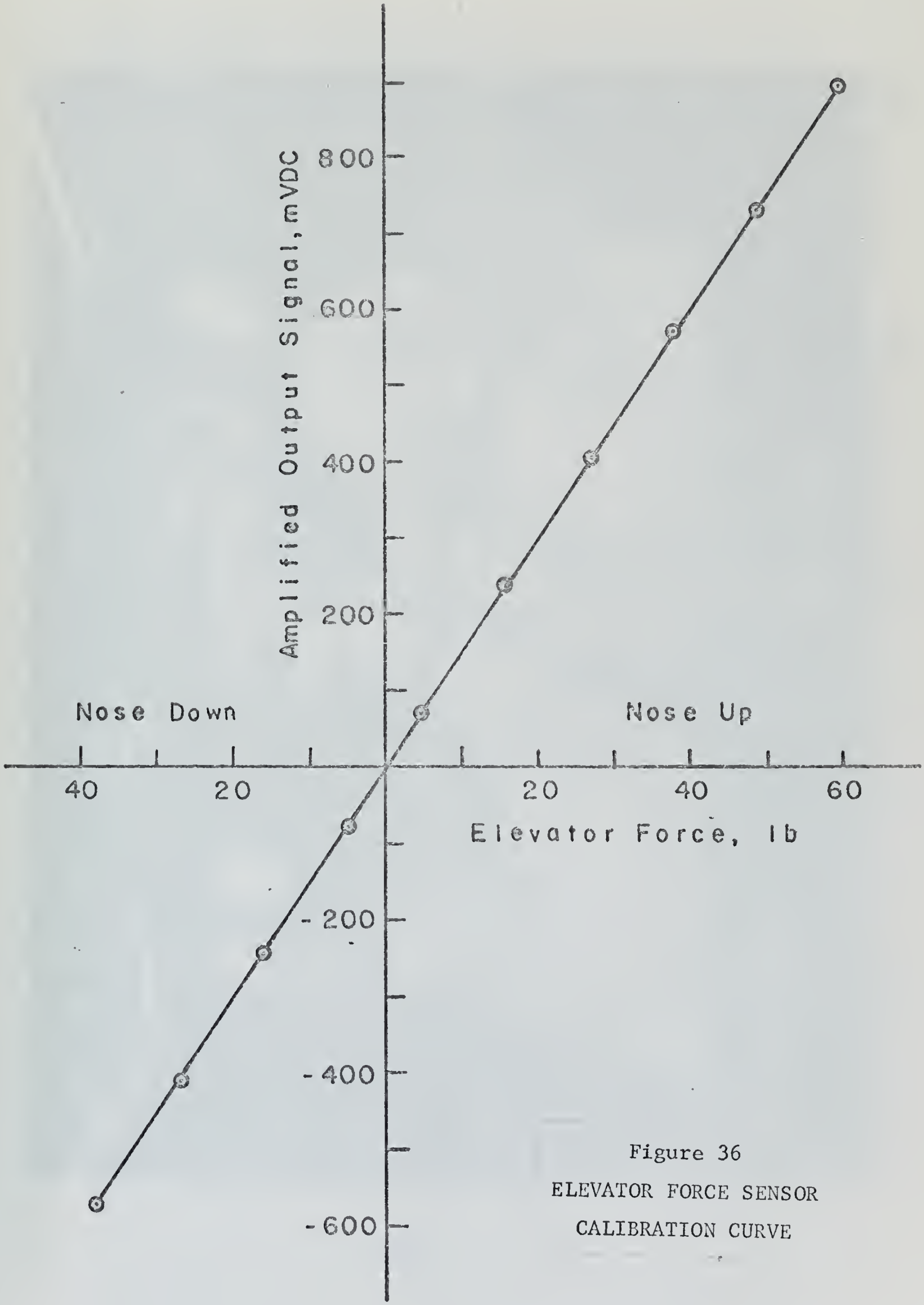


Figure 36
ELEVATOR FORCE SENSOR
CALIBRATION CURVE

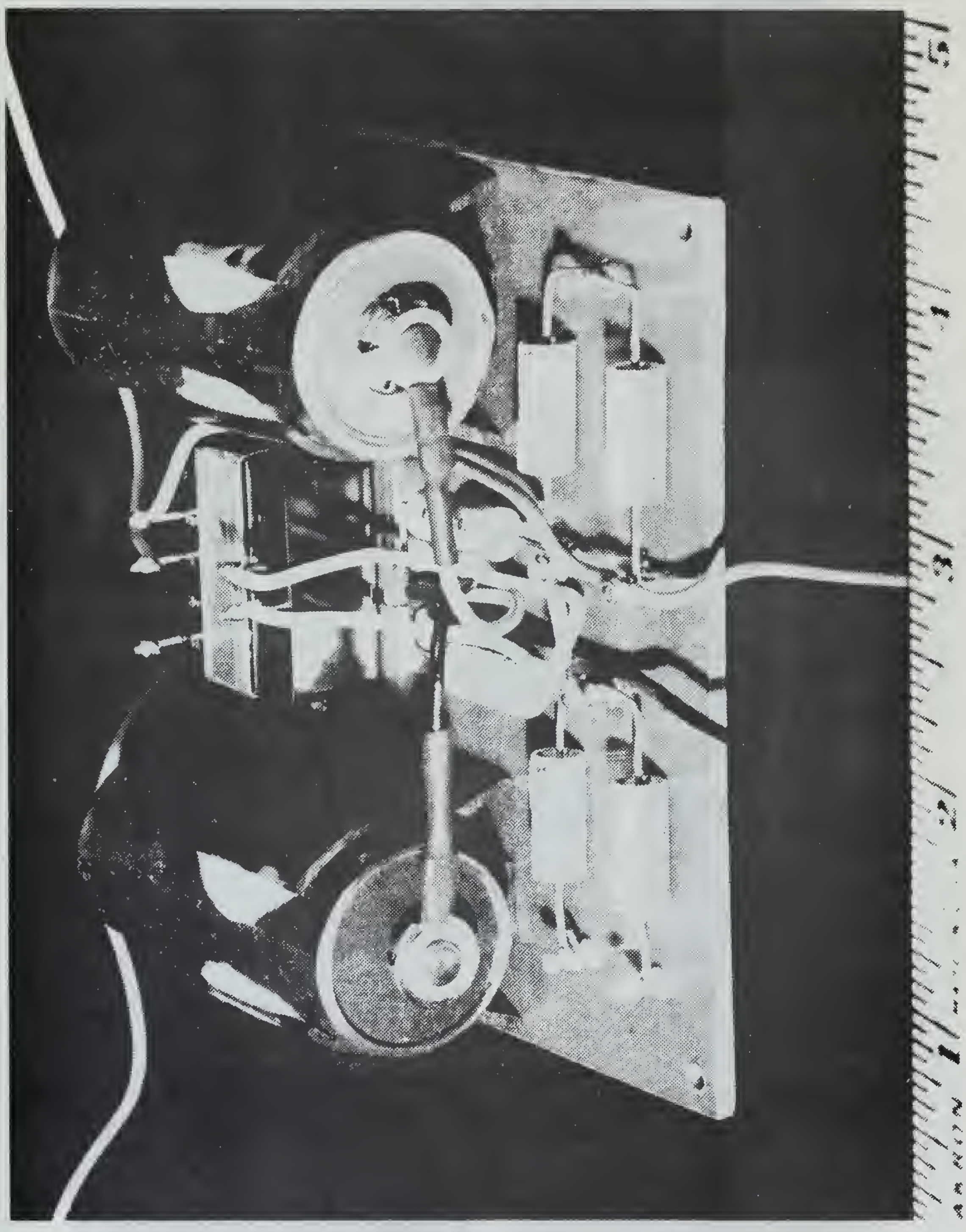


Figure 37

RUDDER PEDAL FORCE

VOLTAGE SUMMER

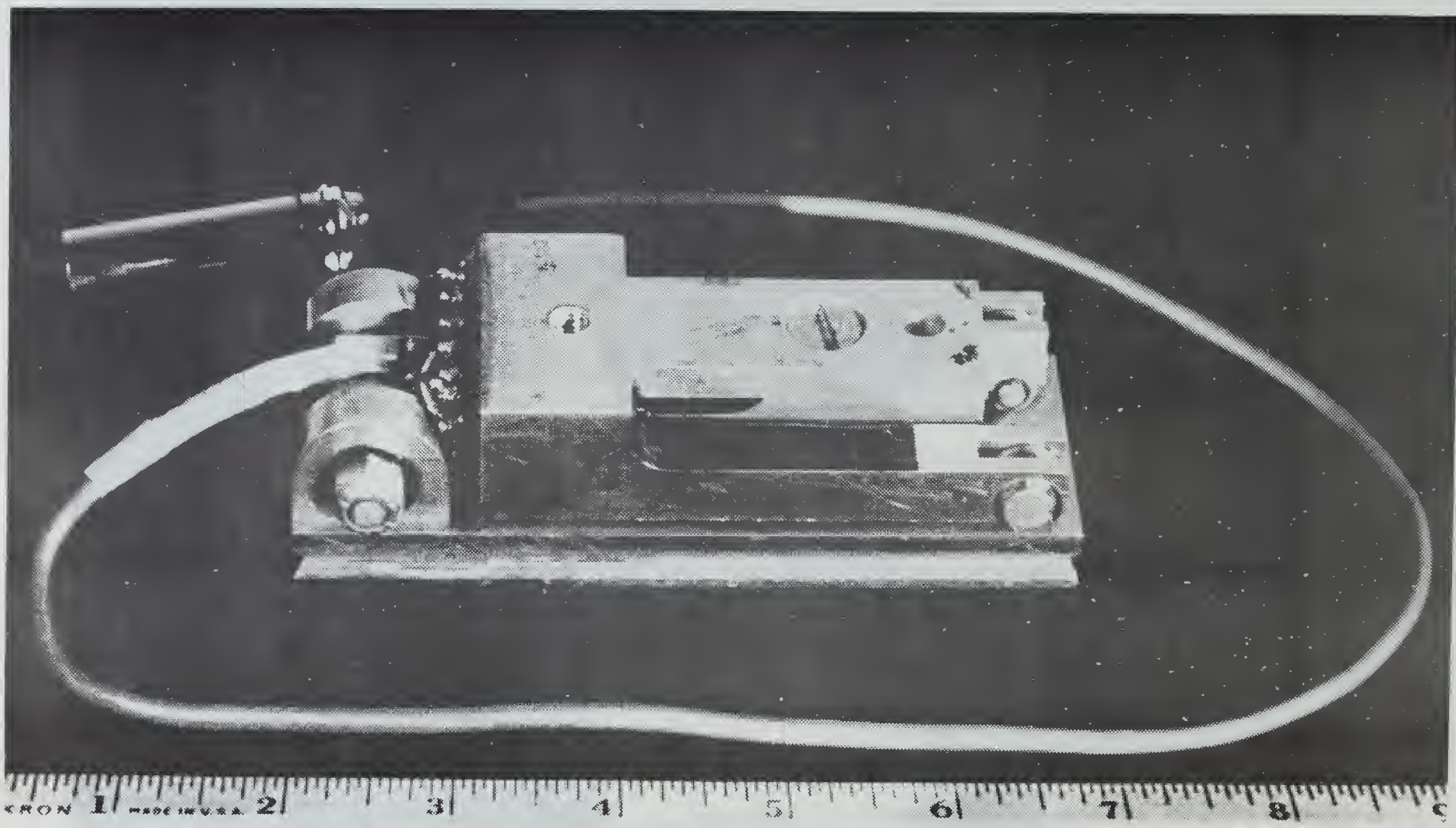


Figure 38

RUDDER PEDAL FORCE TRANSDUCER

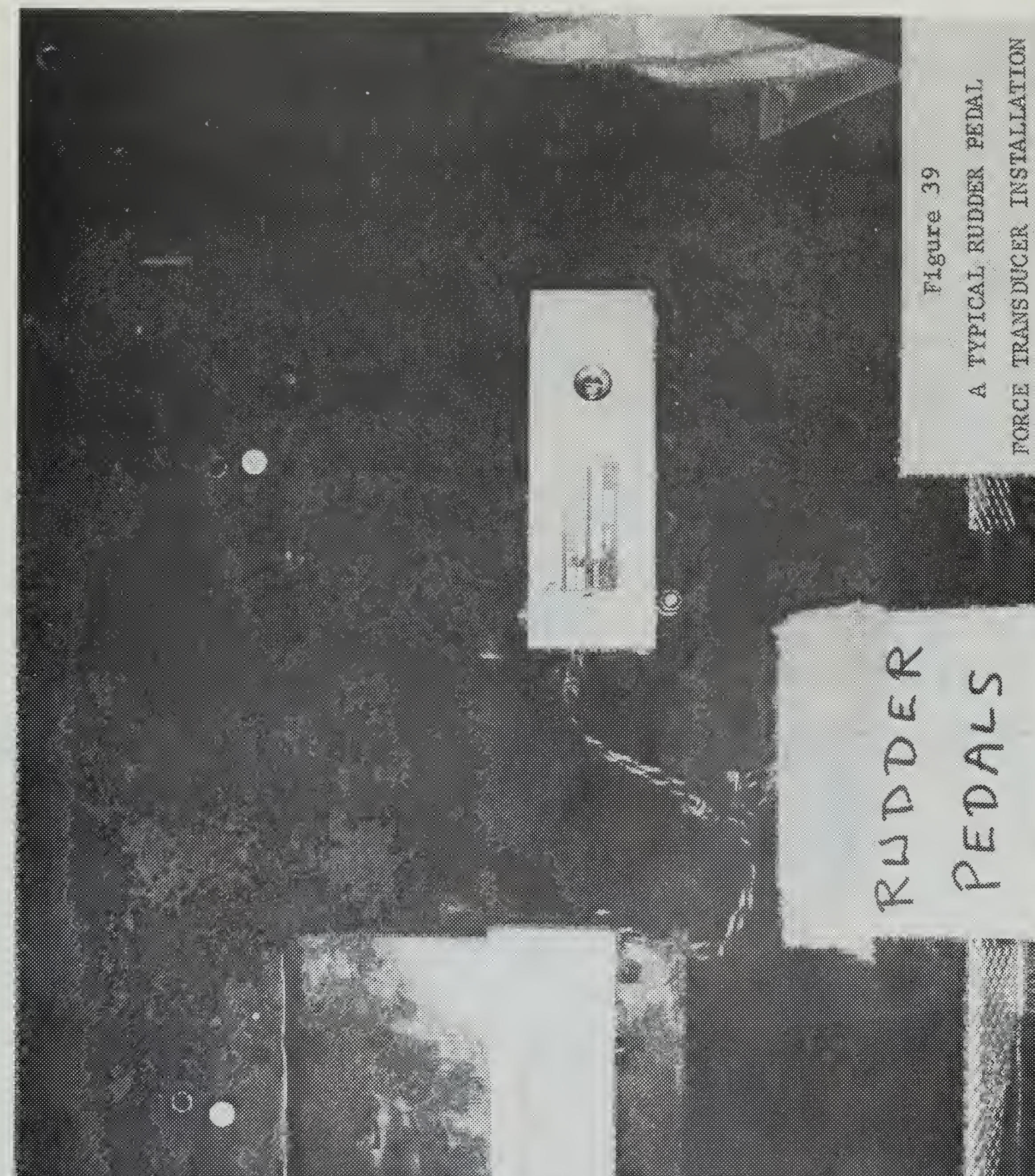


Figure 39

A TYPICAL RUDDER PEDAL
FORCE TRANSDUCER INSTALLATION

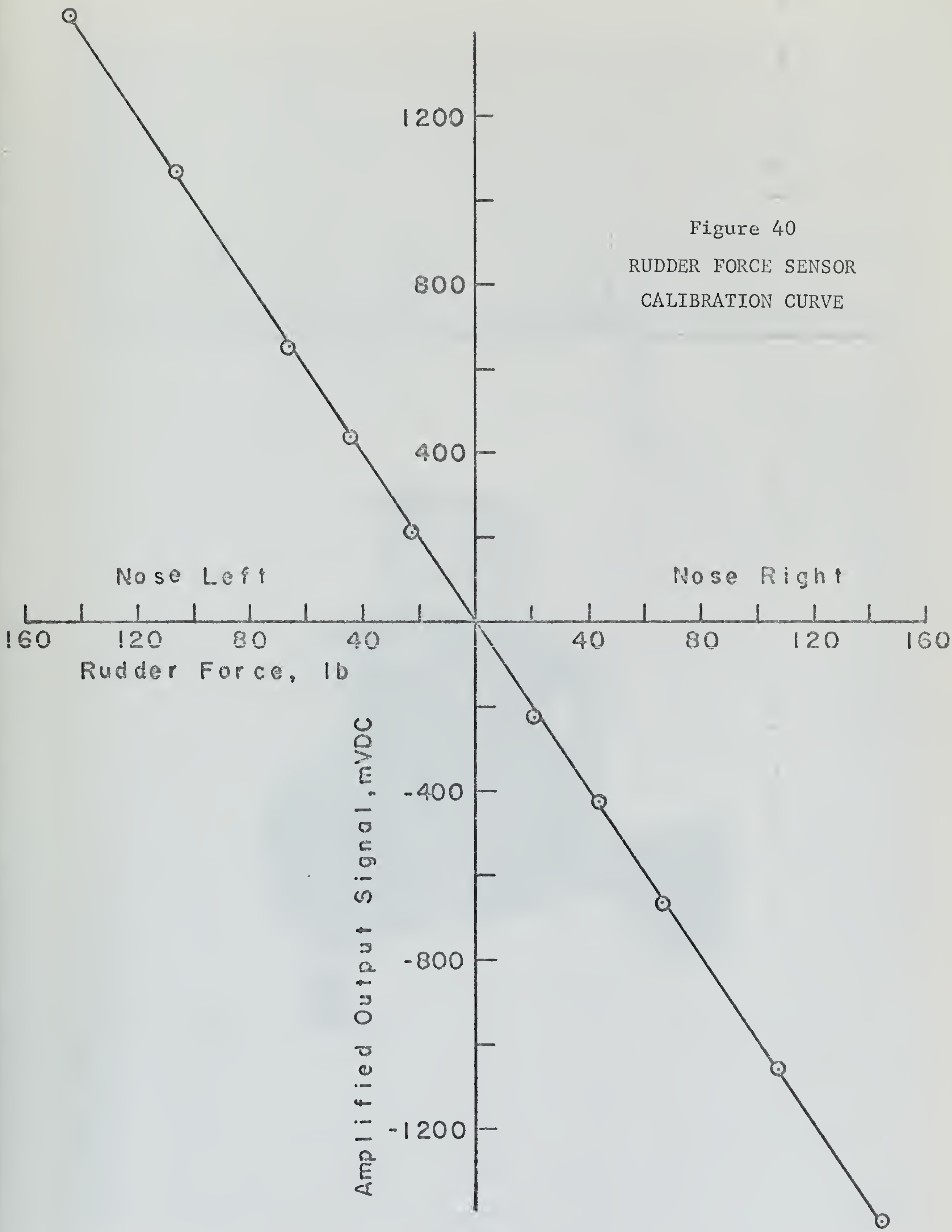


Figure 40
RUDDER FORCE SENSOR
CALIBRATION CURVE

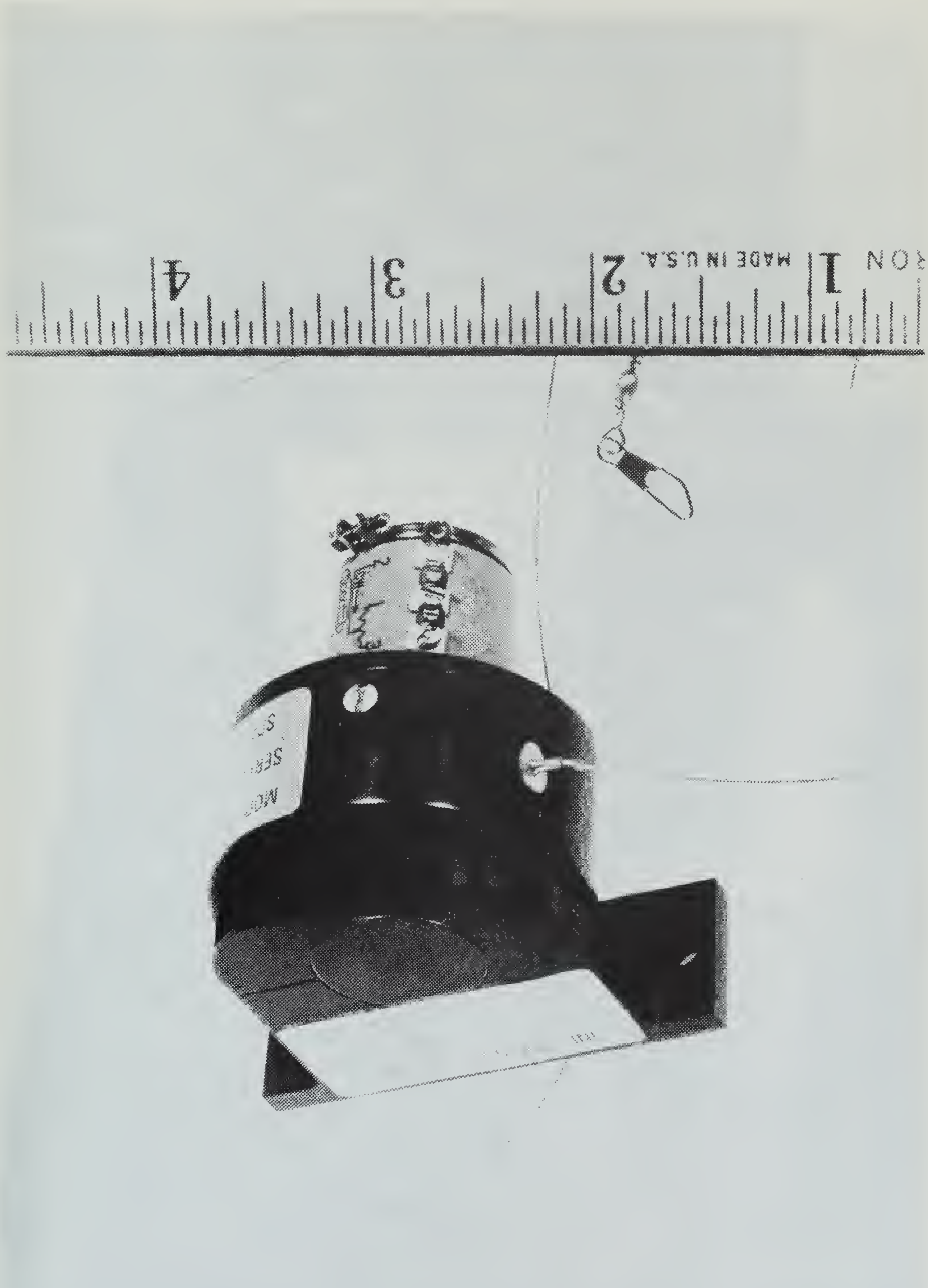


Figure 41

CONTROL SURFACE POSITION LINEAR DISPLACEMENT TRANSDUCER

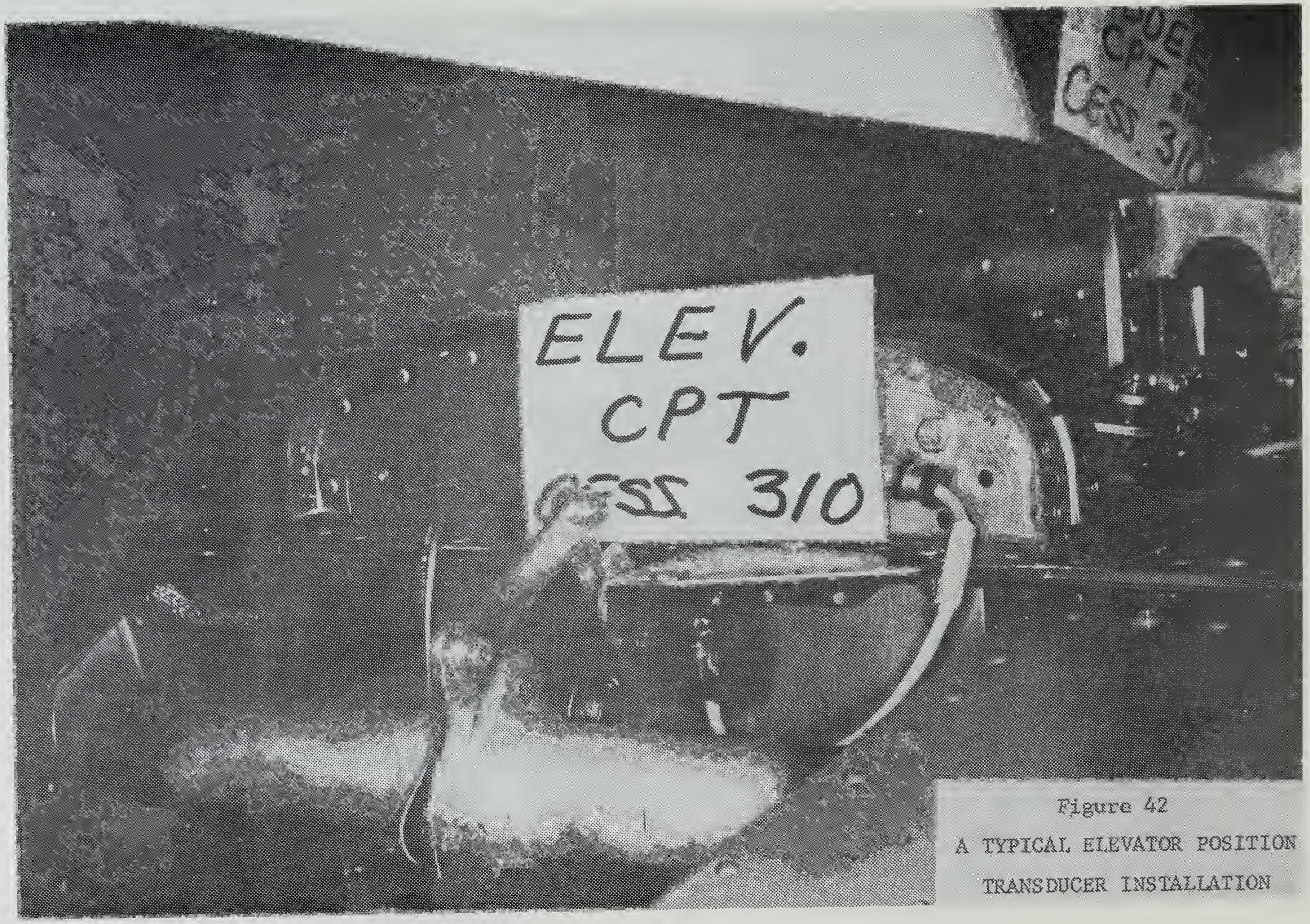


Figure 42
A TYPICAL ELEVATOR POSITION
TRANSDUCER INSTALLATION

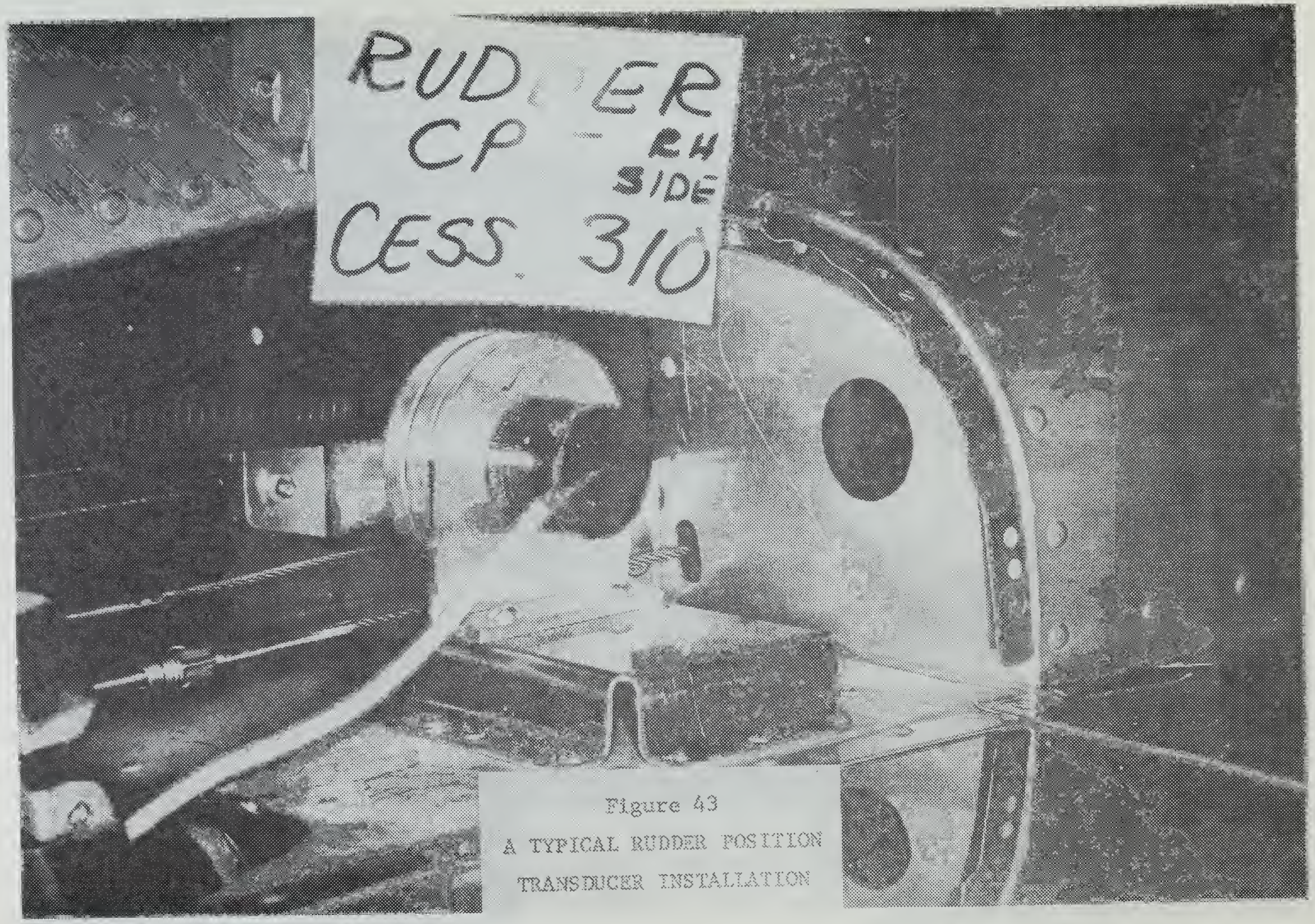


Figure 43
A TYPICAL RUDDER POSITION
TRANSDUCER INSTALLATION

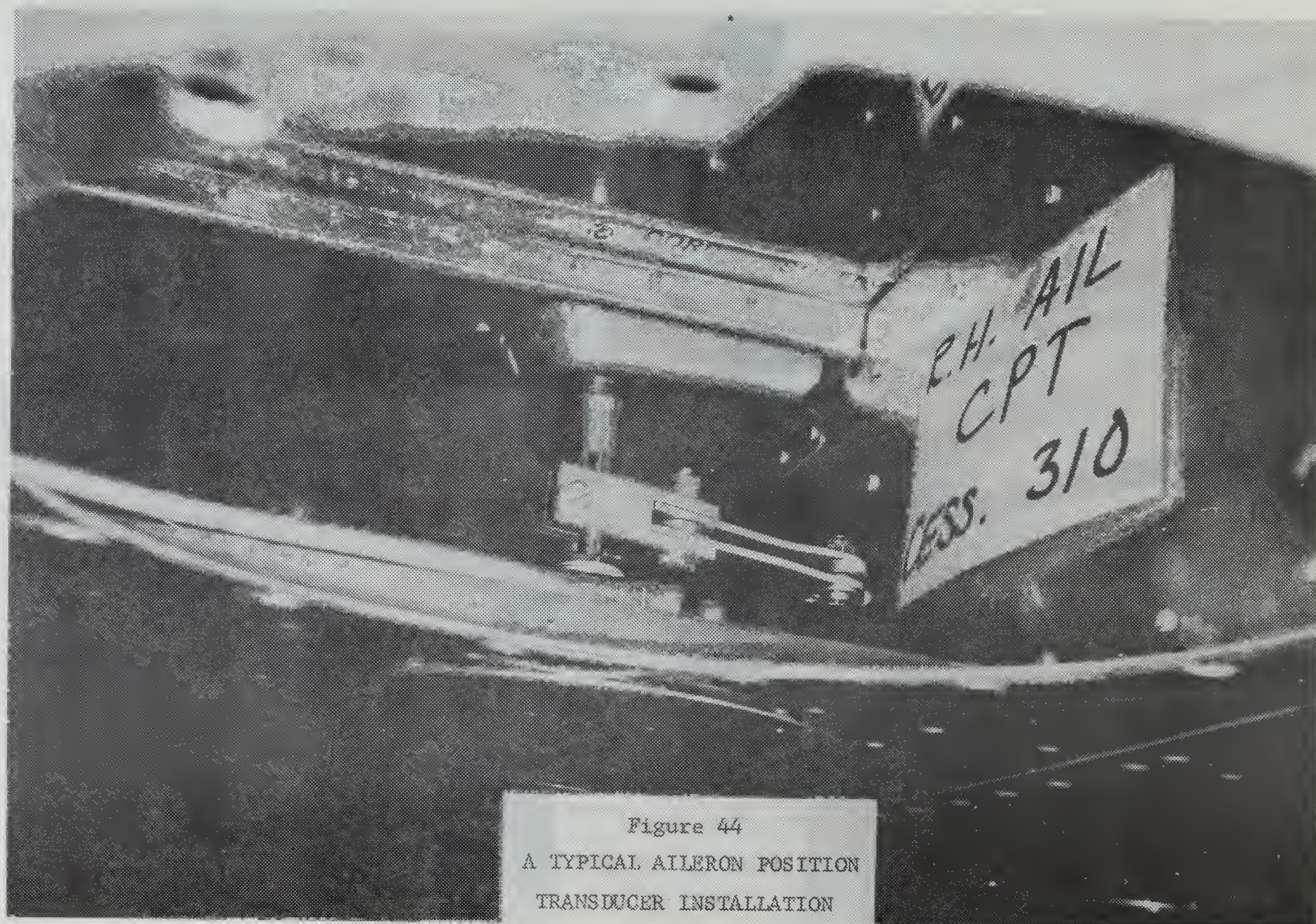


Figure 44
A TYPICAL AILERON POSITION
TRANSDUCER INSTALLATION

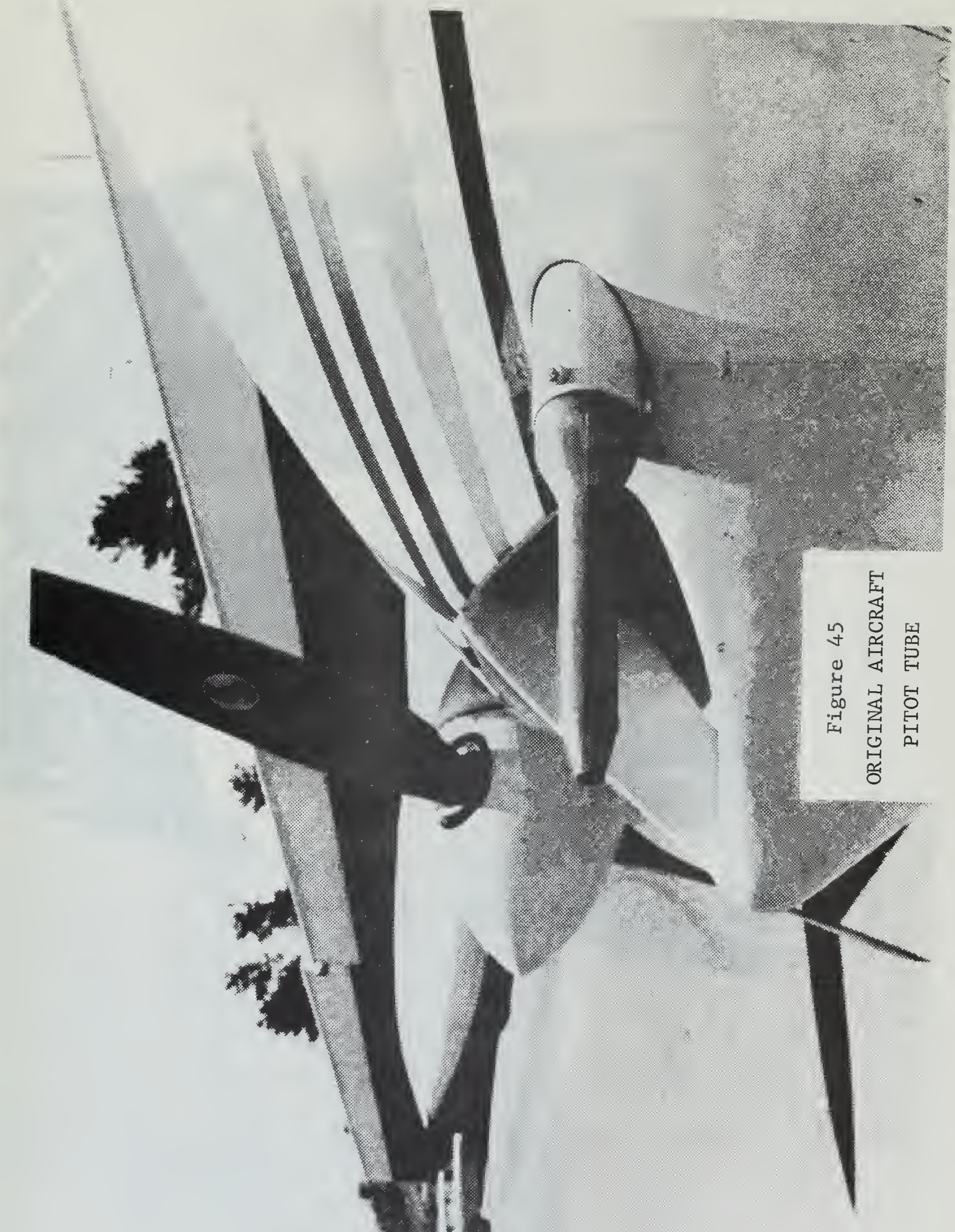


Figure 45
ORIGINAL AIRCRAFT
PITOT TUBE

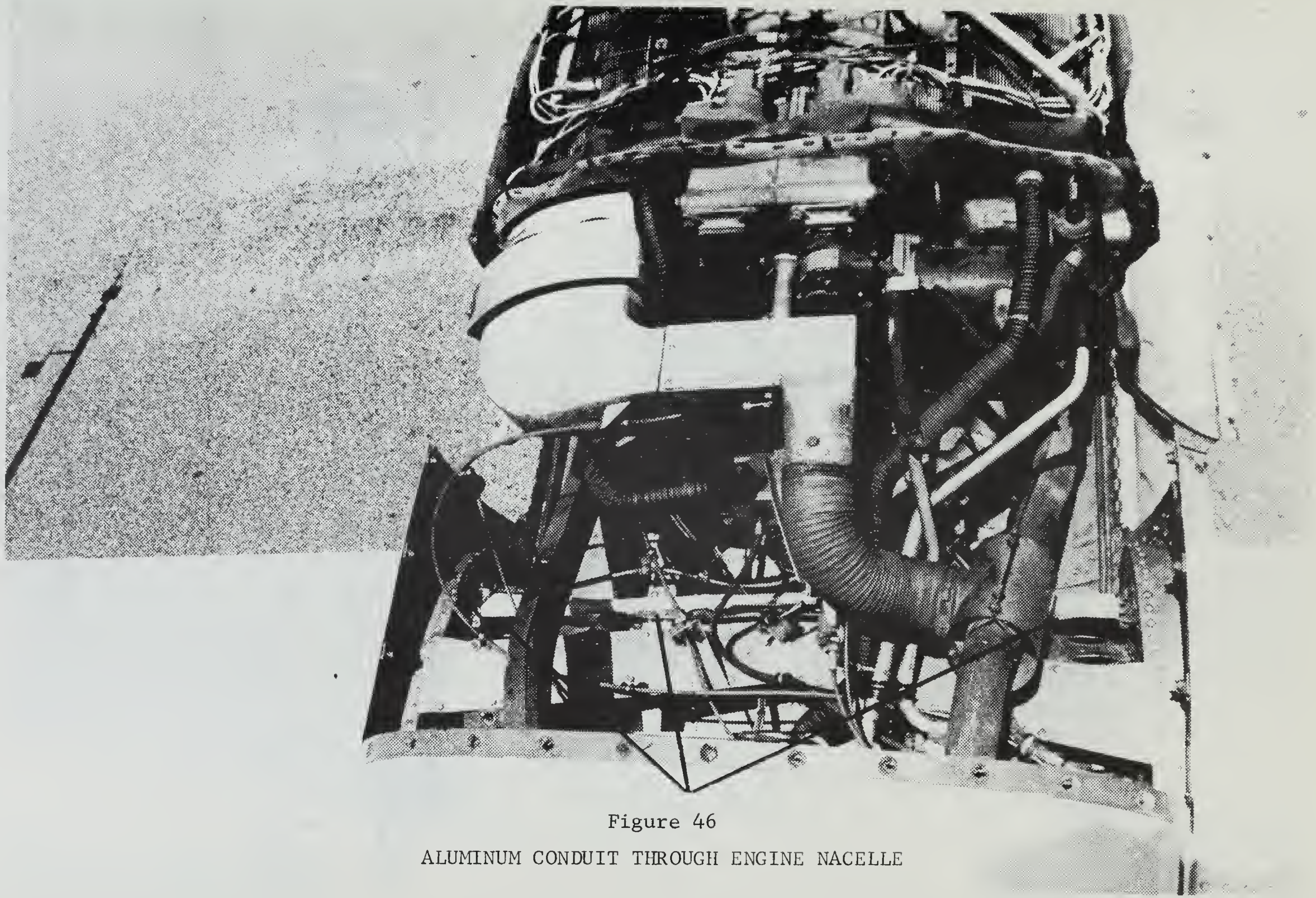


Figure 46

ALUMINUM CONDUIT THROUGH ENGINE NACELLE

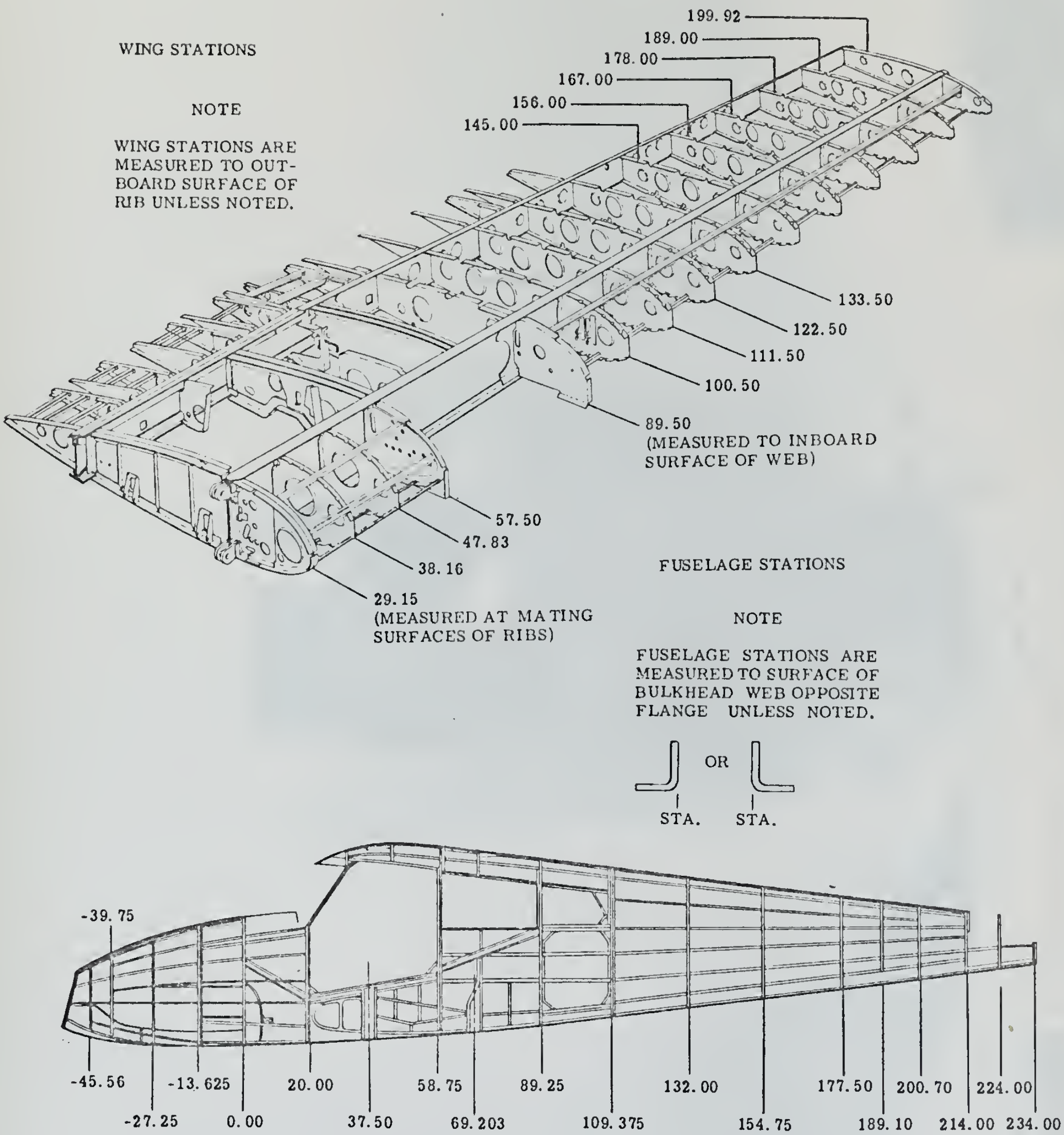


Figure 47

STATIONS AND FRAMES DIAGRAM

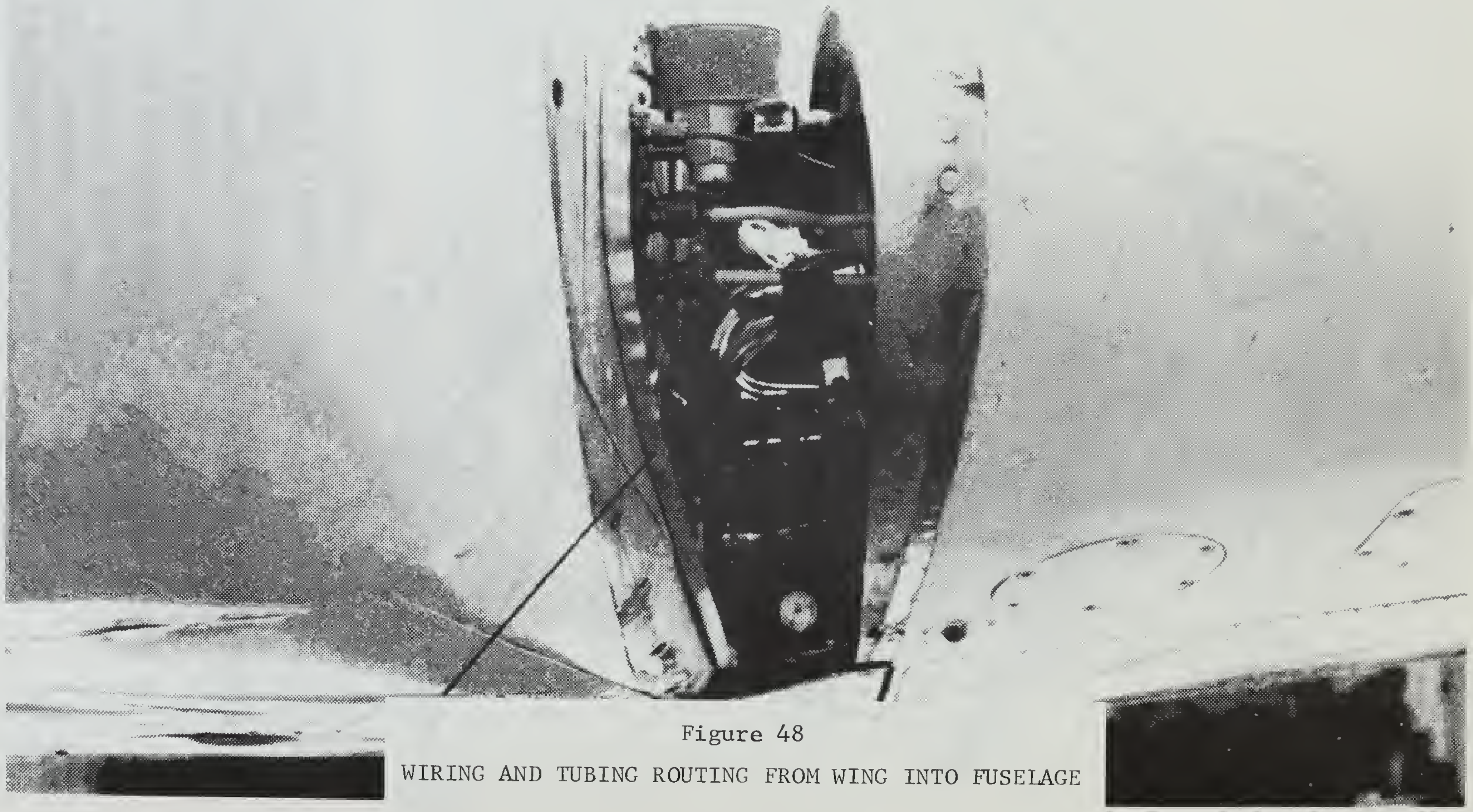
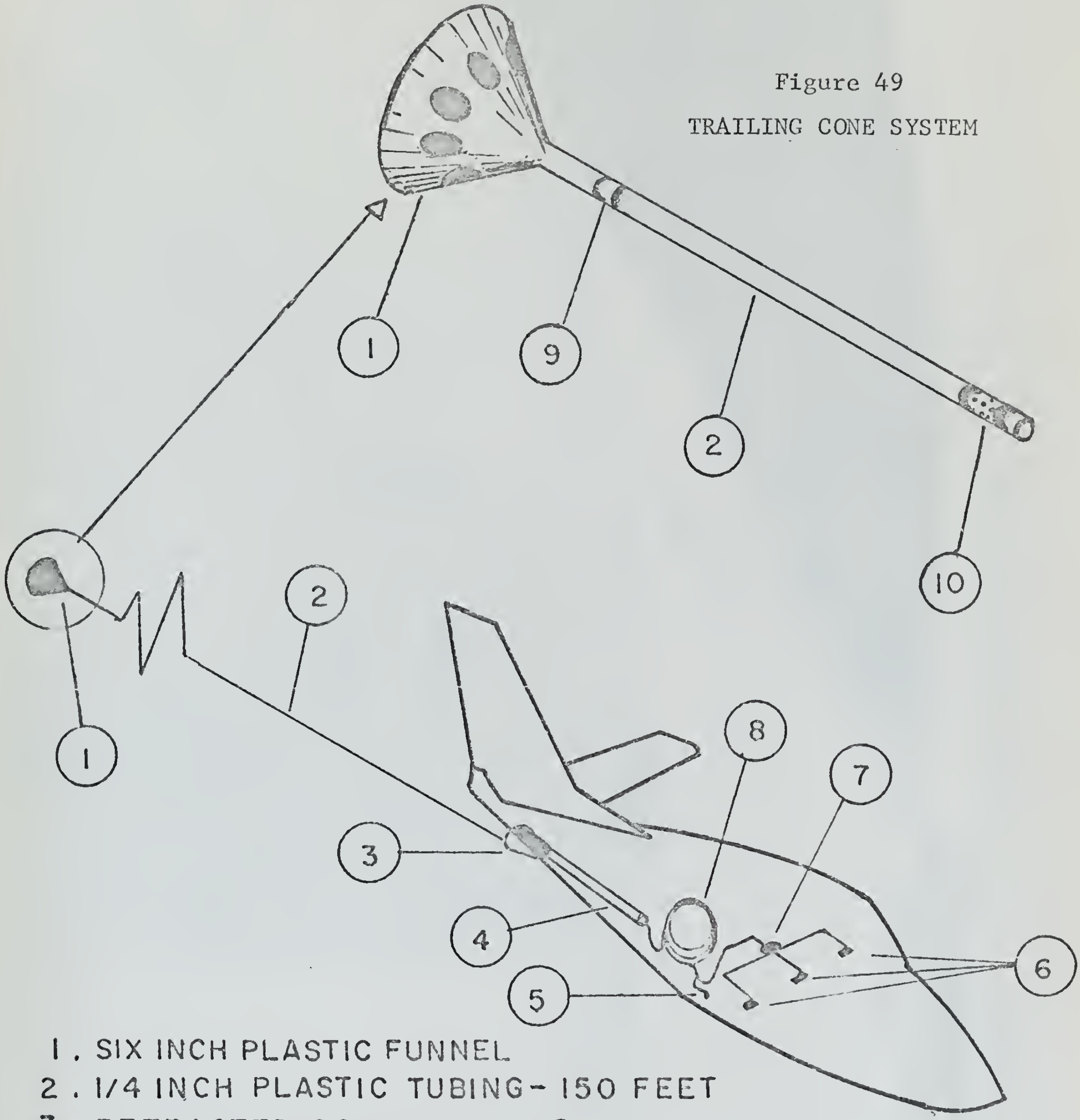


Figure 48

WIRING AND TUBING ROUTING FROM WING INTO FUSELAGE

Figure 49

TRAILING CONE SYSTEM



1. SIX INCH PLASTIC FUNNEL
2. 1/4 INCH PLASTIC TUBING - 150 FEET
3. RETRACTED CONE HOUSING
4. ONE INCH ALUMINUM CONDUIT ENCLOSURE FOR TUBING
5. ATTACHED SAFETY CABLE
6. CONSOLE STATIC PRESSURE SELECTOR VALVES
7. STATIC PRESSURE DISTRIBUTION VALVE
8. RETRACTED TUBING STOWAGE
9. CONE ATTACHMENT FITTING
10. STATIC PRESSURE PORT

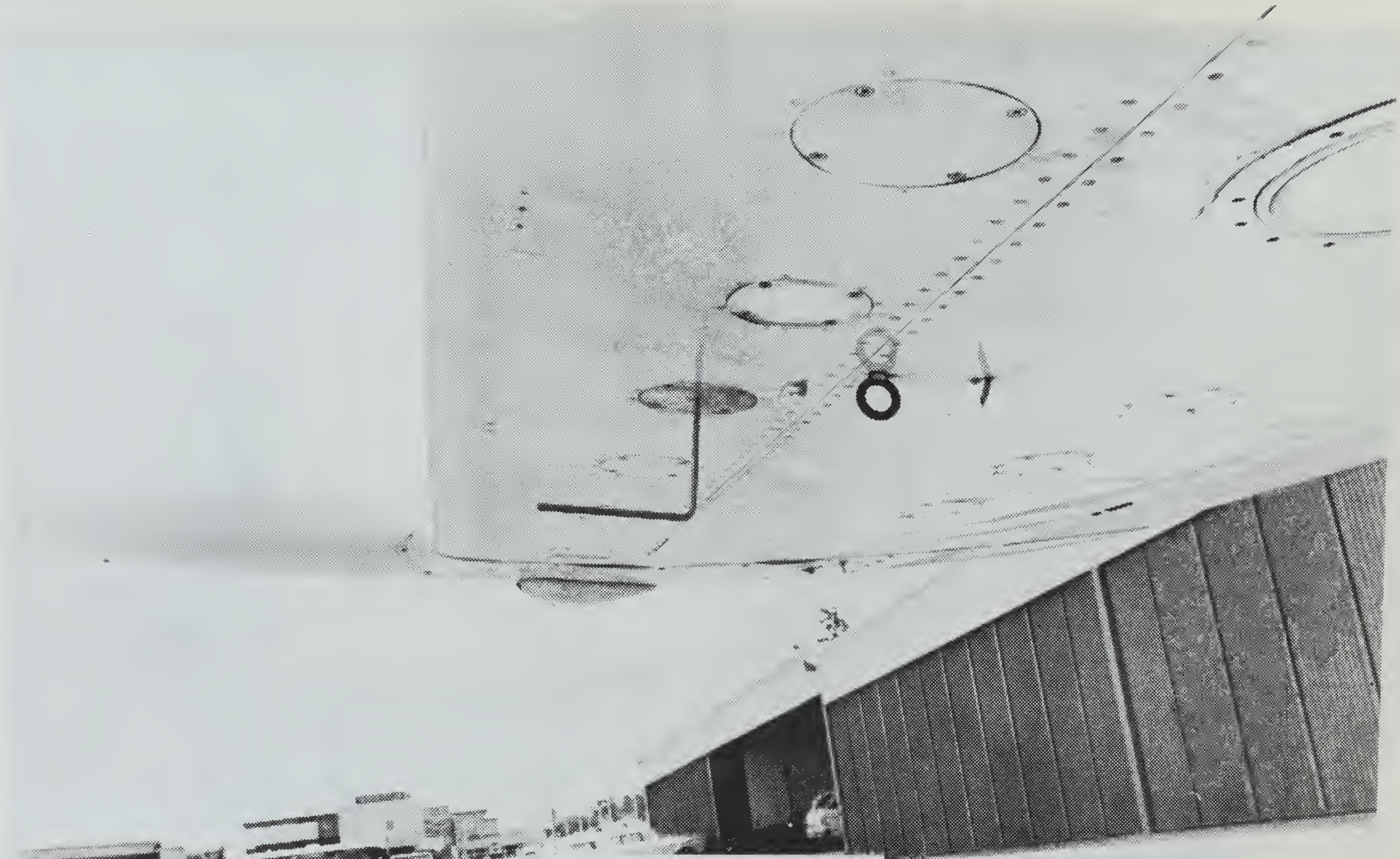


Figure 50

STARBOARD WING PITOT TUBE

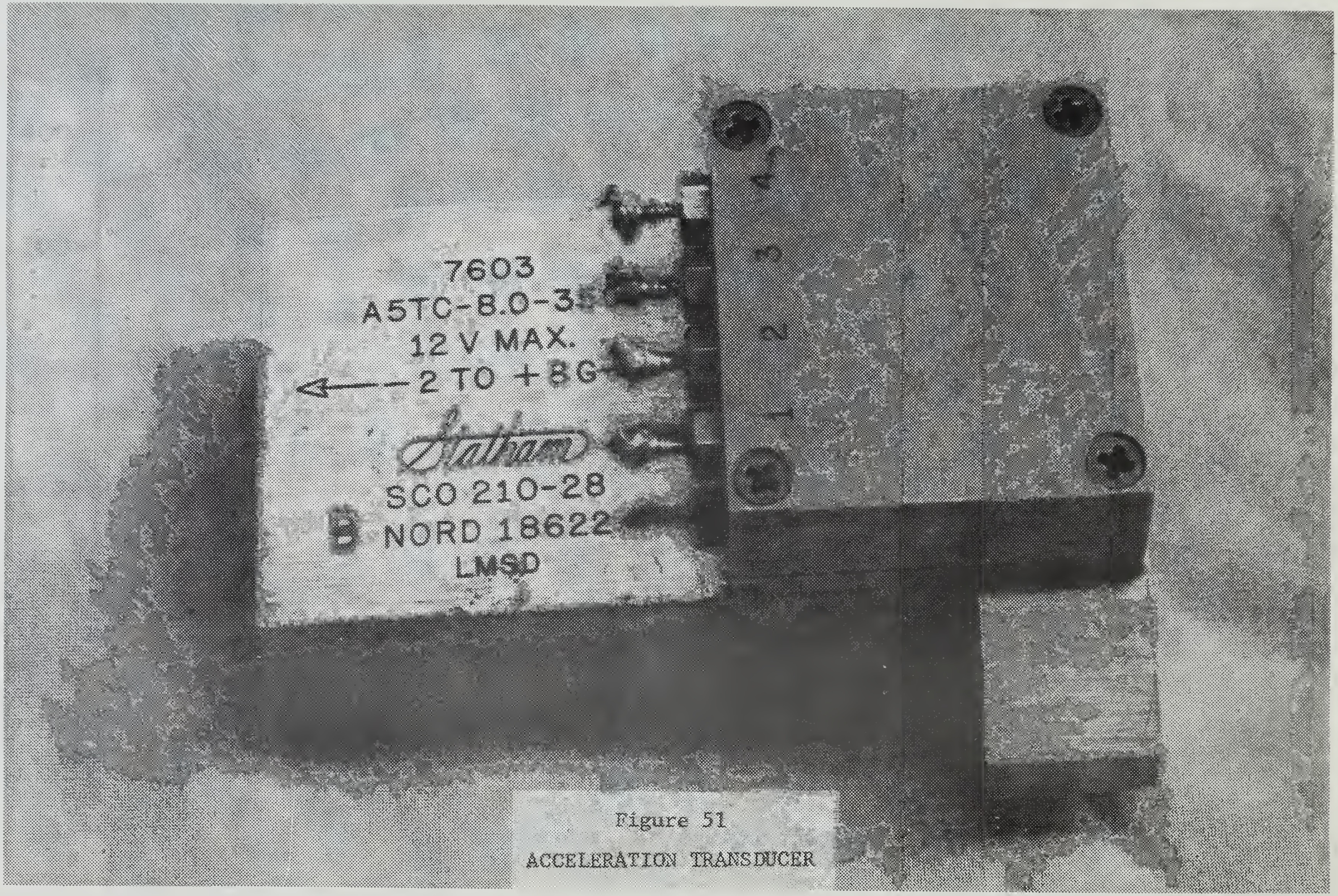


Figure 51

ACCELERATION TRANSDUCER

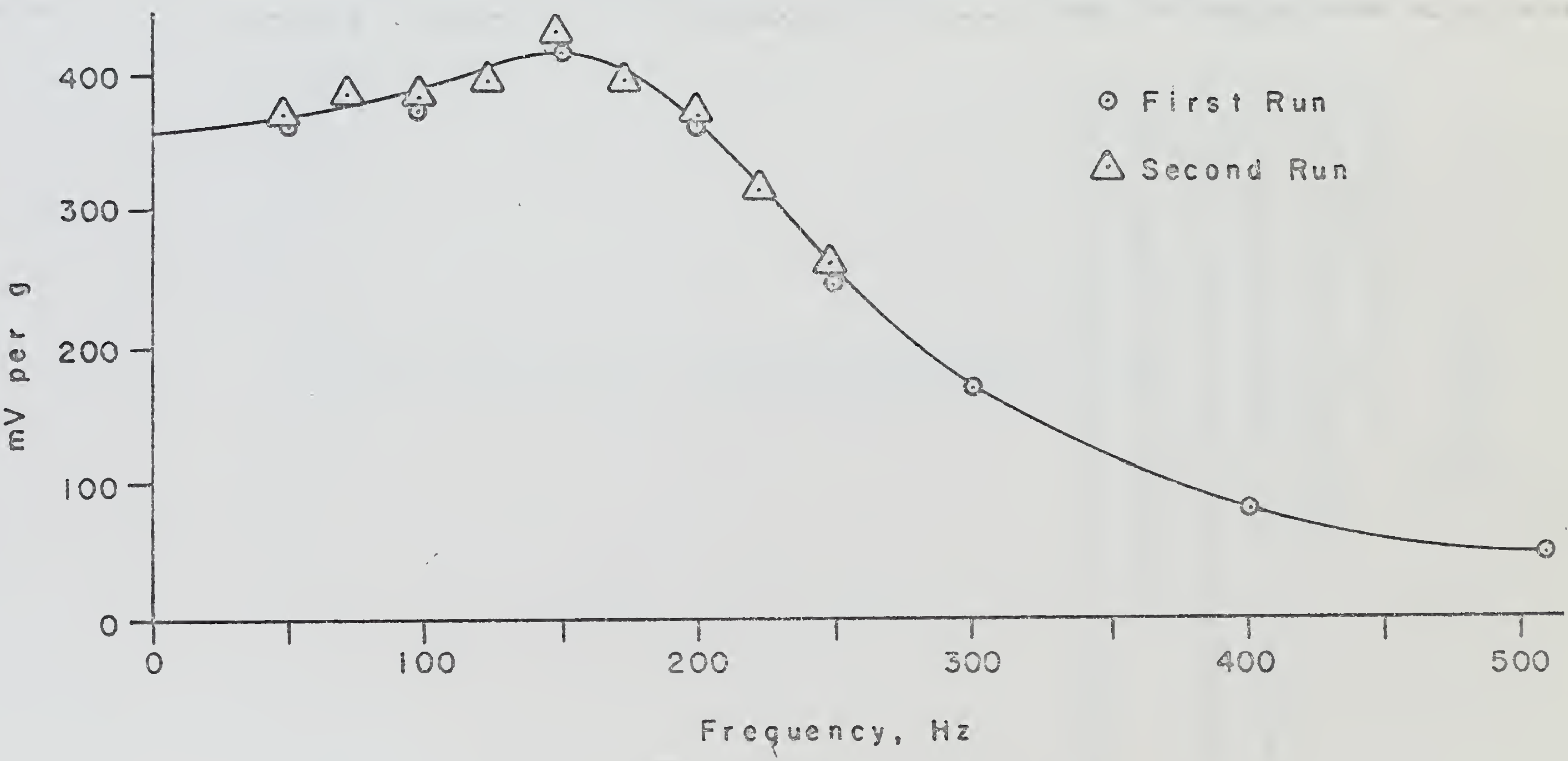


Figure 52
ACCELERATION TRANSDUCER
DYNAMIC CALIBRATION CURVE

APPENDIX B

TABLE I

YAW CALIBRATION

Degrees	Signal Output, Millivolts, dc Nose Left	Nose Right
0	0	0
1	+45	-45
2	+90	-90
3	+134	-134
4	+179	-179
5	+224	-224
6	+268	-268
7	+313	-312
8	+357	-356
9	+402	-400
10	+446	-444
11	+491	-488
12	+535	-532
13	+580	-576
14	+624	-620
15	+669	-664
16	+714	-708
17	+758	-752
18	+803	-796
19	+847	-840
20	+892	-884
21	+937	-928
22	+981	-972
23	+1026	-1016
24	+1070	-1060
25	+1115	-1104
26	+1160	-1147
27	+1204	-1191
28	+1249	-1235
29	+1293	-1279
30	+1338	-1323

TABLE II

ANGLE OF ATTACK CALIBRATION

Degrees	Signal Output, Millivolts, dc Nose Up	Nose Down
0	0	0
1	+43	-44
2	+86	-88
3	+130	-132
4	+173	-176
5	+216	-220
6	+258	-256
7	+300	-302
8	+341	-344
9	+383	-385
10	+425	-426
11	+468	-471
12	+511	-516
13	+554	-560
14	+597	-605
15	+640	-650
16	+686	-695
17	+732	-739
18	+778	-784
19	+824	-828
20	+870	-873

TABLE III

AILERON CONTROL WHEEL FORCE CALIBRATION

Force Pounds	Signal Output, Millivolts, dc Right Wing Down	Left Wing Down
0	0	0
2	-30	+30
4	-60	+60
6	-90	+90
8	-122	+122
10	-153	+153
12	-185	+185
14	-217	+217
16	-248	+248
18	-279	+279
20	-311	+311
22	-342	+342
24	-373	+373
26	-404	+404
28	-436	+436
30	-467	+468
32	-499	+500
34	-530	+532
36	-562	+564
38	-593	+596
40	-625	+628

TABLE IV

ELEVATOR CONTROL WHEEL FORCE CALIBRATION

Force Pounds	Signal Output, Millivolts, dc Nose Up	Nose Down
0	0	0
2	+30	-31
4	+60	-62
6	+89	-93
8	+119	-123
10	+149	-154
12	+179	-184
14	+209	-214
16	+239	-244
18	+269	-273
20	+299	-303
22	+329	-332
24	+360	-362
26	+390	-391
28	+420	-421
30	+451	-450
32	+481	-479
34	+512	-508
36	+542	-537
38	+573	-566
40	+602	-595
42	+632	
44	+661	
46	+690	
48	+719	
50	+749	
52	+778	
54	+807	
56	+836	
58	+865	
60	+894	

TABLE V

RUDDER PEDAL FORCE CALIBRATION

Force Pounds	Signal Output, Millivolts, dc Nose Left (#43)	Nose Right (#105)
0	0	0
5	+50	-48
10	+100	-99
15	+150	-149
20	+200	-199
25	+250	-249
30	+300	-299
35	+350	-350
40	+400	-400
45	+450	-451
50	+500	-500
55	+550	-552
60	+600	-604
65	+650	-651
70	+700	-704
75	+750	-754
80	+800	-805
85	+850	-855
90	+900	-900
95	+950	-950
100	+1000	-1000
105	+1050	-1044
110	+1100	-1094
115	+1150	-1144
120	+1200	-1194
125	+1250	-1241
130	+1300	-1291
135	+1350	-1336
140	+1400	-1386
145	+1450	-1431
150	+1500	-1477
155	+1550	-1526
160	+1600	-1575

TABLE VI

DYNAMIC RESPONSE CALIBRATION OF ACCELEROMETER

Frequency Hertz	Statham Signal Output Millivolts, RMS	Bentley Signal Output Volts, RMs	g's	Millivolts per g
		FIRST RUN		
49.8	192	0.730	0.529	362.9
95.2	221	0.225	0.596	370.8
148.1	211	0.080	0.513	411.3
199.5	181	0.043	0.500	362.0
248.7	138	0.0315	0.569	242.5
299.0	153	0.0325	0.849	180.2
401.1	77.5	0.0215	1.010	76.73
508.1	74.0	0.0215	1.622	45.62
606.8	49.1	0.0185	1.990	47.6
		SECOND RUN		
47.0	202	0.855	0.552	365.9
71.6	195	0.350	0.524	385.5
97.3	202	0.192	0.531	380.4
123.1	200	0.115	0.509	392.9
146.7	194	0.072	0.452	429.2
173.6	183	0.053	0.467	391.86
198.5	168	0.0398	0.458	366.8
221.4	148	0.033	0.473	312.9
247.0	126	0.027	0.481	261.9

Frequency Hertz	Phase Lag Degrees
49	13.7
73	21.3
98	26.5
122	35.0
149	51.0
173	66.0
198	78.0
217	92.0
249	108.0
299	127.0

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13. ABSTRACT

The two-course study of flight evaluation techniques offered by the Aeronautics Department of the Naval Postgraduate School requires an airborne laboratory phase of instruction that introduces the student to the actual problems encountered in obtaining accurate in-flight data. To satisfy this need, a civilian registered Cessna 310H aircraft, N164X, was leased by the Naval Postgraduate School in April, 1973. An airborne data acquisition system was designed and installed that allows three students to obtain individual measurements of twelve performance and stability and control parameters. The measurements are obtained using both electrical and differential pressure sensors, and are manually recorded by each student. Due to time constraints, no in-flight evaluation of the system has yet been conducted.

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
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INSTRUMENTATION						
DATA ACQUISITION						
FLIGHT CONTROLS						
VANE						
CLASSROOM						
CONSOLES						

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